Small Satellite Research Center
United States Air Force Academy

USAFA-FS2-GSDP-23
Revision 2
October 2002

GROUND SAFETY DATA PACKAGE

FALCONSAT-2

Small Satellite Research Center
Department of Astronautics
U.S. Air Force Academy, CO 80840

SIGNATURES

Customer Payload: FS-2 Spacecraft

Customer: U.S. Air Force Academy (USAFA), Space Test Program (STP)

Date: October 2002

Customer Approval:

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REVISIONS

Revision	Description	Date	Approval
1	First Turn-in Jun 2002	6/04/02	
2	Final Revision	10/28/02	

1. INTRODUCTION

1.1 Purpose

This document comprises the Phase 2/3 Ground Safety Data Package (GSDP) for the FalconSat-2 (FS-2) satellite being developed by the United States Air Force Academy (USAFA). FS-2 is manifested to fly aboard the shuttle for deployment as part of the Hitchhiker program. FS-2 will be integrated into a Hitchhiker (HH) canister using the Pallet Ejection System (PES). The canister will have a Hitchhiker Motorized Door Assembly (HMDA) opening lid. FS-2, with its Miniature Electrostatic Analyzer (MESA) experiment, is a nano-satellite designed to investigate ionospheric plasma depletions that cause radio transmission disruptions by conducting *in situ* sampling of plasma density.

FS-2 is designed to be inactive while installed in the Shuttle. This is ensured by the use of four, series-connected micro-switches that inhibit power from the solar panels to the spacecraft batteries while in the launch configuration. Once deployed from the Shuttle, FS-2 will begin operation by initializing its hardware, charging its batteries, and preparing for contact with USAFA's ground station after a minimum of three orbits.

The purpose of this GSDP is to provide the Shuttle program a comprehensive safety assessment of FS-2 subsystems that potentially pose hazards during ground testing and integration. This GSDP is intended to enumerate all potential hazards associated with the FS-2 payload and provide sufficient information to show that FS-2 complies with the requirements specified in NSTS 1700.7.

1.2 Scope

This GSDP encompasses all subsystems associated with the FS-2 spacecraft as a Shuttle secondary payload. FS-2 will use hitchhiker government furnished equipment (GFE). NASA GFE includes the PES adapter ring, hitchhiker canister, canister heaters and base plate thermistor, avionics package, and Pallet Ejection System. FS-2 will be mounted inside the hitchhiker canister via the PES. NASA/GSFC will provide the mating launch vehicle interface (LVI) with the prescribed dimensional configuration as defined in the *Hitchhiker Customer Accommodations & Requirements Specification,* NASA/GSFC 740-SPEC-008, 1999, pp. 1-250 thru 2-148.

The FS-2 payload is divided into five subsystems plus ground support equipment (GSE). The FS-2 subsystems include:

- Structure
- Electrical Power Subsystem (EPS)
- Communication Subsystem (COM)
- Data Handling Subsystem
- Miniature Electrostatic Analyzer (MESA) experiment

The FS-2 GSE will be used exclusively at the hitchhiker integration sites, both at NASA/GSFC and at NASA/KSC and may be required at non-Hitchhiker sites such as the OPF or pad (battery charging only). The FS-2 GSE will be used to accomplish functional and verification testing; charge/discharge and recondition the flight battery; and to verify status of safety inhibits. Specific operations are discussed later in this GSDP.

1.3 Customer Data

FS-2 customer data are listed in Table 1.

Table 1: Customer Data

Customer Payload Name	FalconSAT-2 Spacecraft
Customer Payload Acronym	FS-2
Customer Information	Small Satellite Research Center Department of Astronautics US Air Force Academy, CO, 80840
Program Manager (USAFA)	LtCol Jerry Sellers 719-333-3315, Jerry.Sellers@ usafa.af.mil
DoD Payload Manager (SMC/STH)	Maj Don Hill 281-483-3425, don.e.hill1@jsc.nasa.gov
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MESA Experiment Manager	LtCol Francis Chun 719-333-2601, Francis.Chun@usafa.af.mil
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Electrical Systems Engineer (Consultant)	Jim White of Colorado Satellite Services 303-840-1907, jim@coloradosatellite.com
Structural Engineer (Consultant)	Thomas P. Sarafin of Instar Engineering and Consulting, Inc. 303-973-2316, TPSarafin@aol.com

1.4 Safety Analysis

Established analysis methods will be used qualitatively to show the likely hazards associated with the FS-2 subsystems. Operational conditions of the FS-2 payload will be described in detail to identify potential safety hazards and their associated countermeasures, or inhibits, implemented in the design.

1.5 Safety Status Summary

The FS-2 design is 100% complete. The FS-2 payload has completed the Phase 0/1 safety review and is preparing for a phase 2/3 safety review.

1.5.1 Action Items

There are no action items at this time pending review by the Payload Safety Review Panel.

1.5.2 Noncompliance Reports (NCRs)

There are no noncompliance reports at this time, nor are any NCRs anticipated for the FS-2 payload.

1.5.3 Operational Controls Identification

None of the hazards identified will need to be controlled operationally.

1.6 Applicable Documents

FalconSAT-2 Battery Charging and Re-conditioning Procedures (FS2-BATTPROC-23)

2. PAYLOAD DESCRIPTION

2.1 Mission Objectives

There are three main objectives for the FalconSAT-2 (FS-2) mission:

- The primary science objective is to investigate ionospheric plasma depletions that cause radio transmission disruptions. This will be accomplished using the miniature electrostatic analyzer (MESA) instrument onboard the FalconSAT-2 spacecraft and represents a DoD Space Experiments Review Board (SERB) mission.
- The overall programmatic objective is to provide an opportunity for USAF Academy Cadets to "learn space by doing space," allowing them to participate in all phases of mission design, assembly, test, and operations.
- A final, longer-term program objective is to validate key technologies, design concepts and processes that can be used for follow-on FalconSAT missions.

2.2 Physical Description

The primary structure for the FS-2 microsatellite is a 12.5 in (31.75 cm) cube. With antennas and PES adapter ring, the total height is 20.9 in (53.1 cm). The total mass is 43.25 lbs. The external configuration is shown in Figure 1. The majority of FS-2 components will be mounted inside the primary structure. FS-2 subsystems include:

- Structure
- Electrical Power Subsystem (Power Conditioning Unit, Battery, Solar Panels) (EPS)
- Data Handling System
- Communication Subsystem (VHF Rx, S-Band TX) (COMM)
- Thermal Control
- Attitude Determination and Control Subsystem (ADCS)
- MESA Experiment

An expanded view of the spacecraft is shown in Figure 2. Physical characteristics of the subsystems, along with operating temperature ranges are shown in Table 2. The complete functional description for each of the FS-2 subsystems is presented in Section 2.3.

FS-2 will be mounted in a HH Canister (with heating capability), with Hitchhiker Motorized Door Assembly (HMDA) and will use the PES.

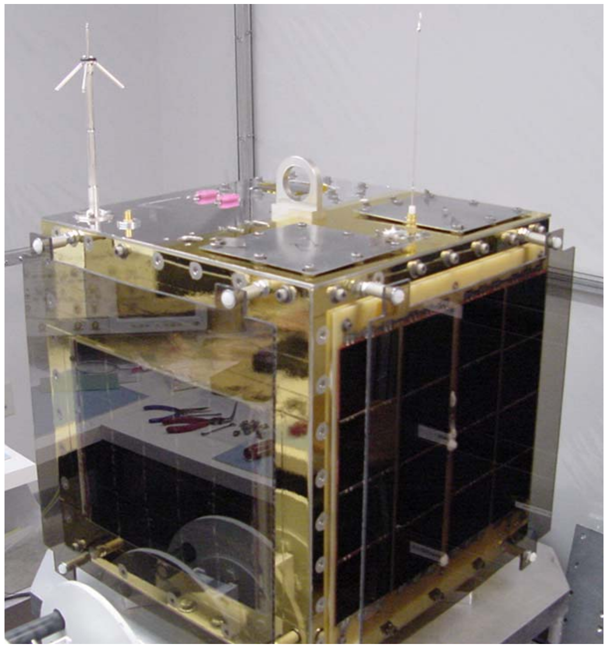


Figure 1: FS-2 external view

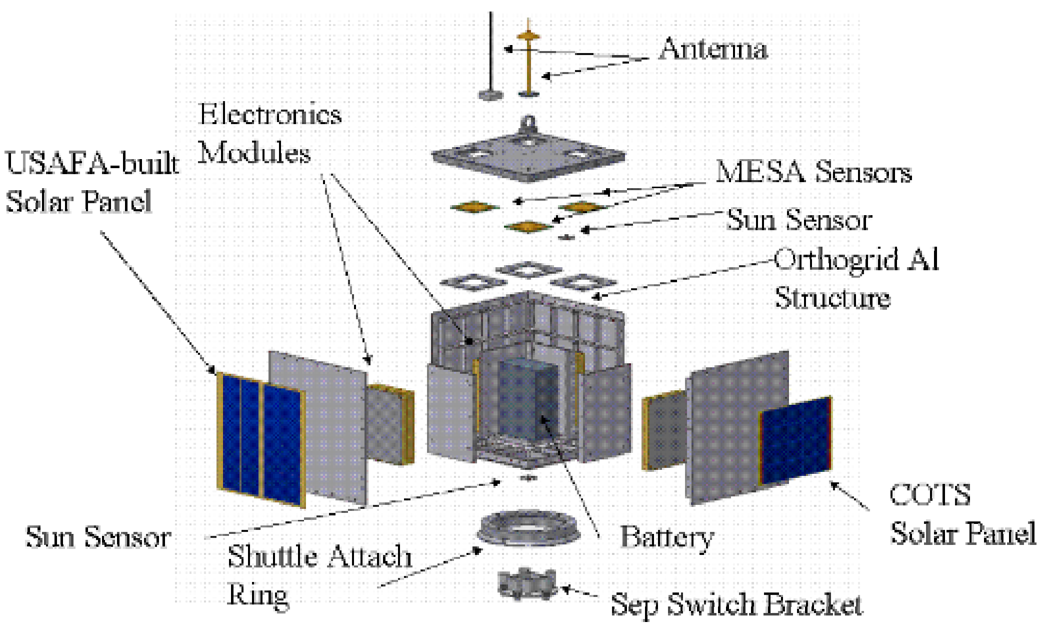


Figure 2: FS-2 nanosatellite showing key features and components.

Table 2: Summary of FS-2 Subsystem Characteristics

Item	**Payload Characteristic**	*MESA*	*Structure*	*EPS*	*Data Handling*	*Comm*	**FS-2**
1	Weight (approx.)	1 lbs	35 lbs	5 lbs	2 lbs	2 lbs	43.25 lbs
2	Field of View	30°	N/A	N/A	N/A	N/A	30°
3	Nominal Operating Temp (Min)	0° C	N/A	0° C	0° C	0° C	0° C
4	Nominal Operating Temp (Max)	30° C	N/A	30° C	30° C	30° C	30° C
5	Non-Operating Temp (Min)	-40° C	N/A	-30° C	-40° C	-40° C	-30° C
6	Non-Operating Temp (Max)	85° C	N/A	50° C	85° C	85° C	50° C
7	Storage Temp (Min)	-40° C	N/A	-30° C	-40° C	-40° C	-30° C
8	Storage Temp (Max)	85° C	N/A	50° C	85° C	85° C	50° C
9	Safety Temp (Min)	-40° C	N/A	-30° C	-40° C	-40° C	-30° C
10	Safety Temp (Max)	85° C	N/A	100° C	85° C	85° C	50° C

Note for Table 2

1. Battery is the driver for the min/max temperature requirement.

2.3 Subsystem Functional Descriptions

FS-2 hardware is outlined in the following sections:

- Structures (Section 2.3.1)
- Electrical Power System (Section 2.3.2)
- Data Handling (Section 2.3.3)
- Communications (Section 2.3.4)
- Thermal (Section 2.3.5)
- Attitude Determination and Control (ADCS) (Section 2.3.6)
- MESA (Section 2.3.7)

2.3.1 Structure

FS-2 primary structure consists of four milled aluminum side panels, four milled center column plates, a milled base plate, and a milled top panel. All of these components are machined from 6061-T651 and 6061-T6 Aluminum. The spacecraft electronics modules are mounted around the central column walls. The battery box is mounted to the base plate at the center of the column. The top wall will house the sensor arrays, antennas, and a ground handling fixture.

The PES adapter ring is mounted to the base plate and forms half of the PES separation system. The adapter ring was supplied by GSFC. A fit check was performed by GSFC personnel. The base plate will also provide mounting locations for the FS-2 separation micro-switches that will inhibit activation of the spacecraft prior to deployment as well as provide indication of separation from PES. No independent confirmation of separation from the PES is provided by the FS-2 systems.

It is not critical where FS-2 is located in the Payload Bay (PLB). No specific orientation of FS-2 about the PES base plate normal is required.

2.3.2 Electrical Power Subsystem

The FS-2 EPS consists of:

- 4 solar panels
- 1 7-cell battery, housed in a vented aluminum box, vents are along spacecraft y axis
- Power regulation, control and distribution in the form of 2 Printed Circuit Boards (PCB) referred to as the battery charge regulator (BCR), the power conditioning module (PCM) and the power distribution module (PDM). These 2 PCBs are housed together in 2 aluminum trays.

Key features of the FS-2 EPS are summarized in Table 3. A detailed description of each of these components follows.

The solar panels will consist of 2 panels constructed at USAFA and 2 commercial-off-the-shelf panels built by SpaceQuest, Ltd. USA.

All panels will use single-junction GaAs cells. The USAFA-built panels consist of 16 5.5 x 6.5 cm cells bonded directly to printed circuit boards (PCBs) using RTV566 adhesive. The PCBs are then attached to the side panels of the spacecraft structure using fasteners and RTV566.

The Spacequest panels consist of 20 4.1 x 4.1 cm cells bonded to aluminum honeycomb with fiberglass face sheets using RTV566. These solar panels are then mounted to the other two side panels of the spacecraft structure, again using fasteners and RTV566. Photographs of both panels are shown in Figure 3. The components of the solar panels are not considered structural items.

Table 3: Summary of FS-2 EPS Characteristics

EPS Parameter	Value
Number of Solar Panels	4
Orbit Average Power	2.5 W
Bus voltage	5 V (regulated) 8.4-10.5 V (unregulated)
Battery Cell Type	N4000DRL
Number of Battery Cells	7
Battery Capacity	4.3 A-hr
Nominal Battery Operating Temperatures	0°C to 30°C

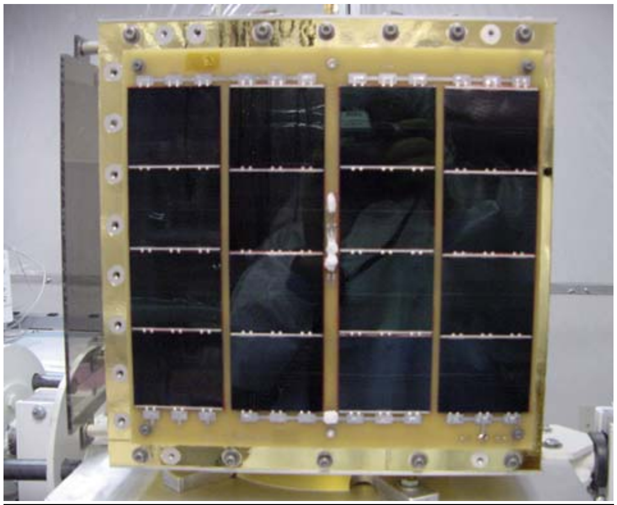
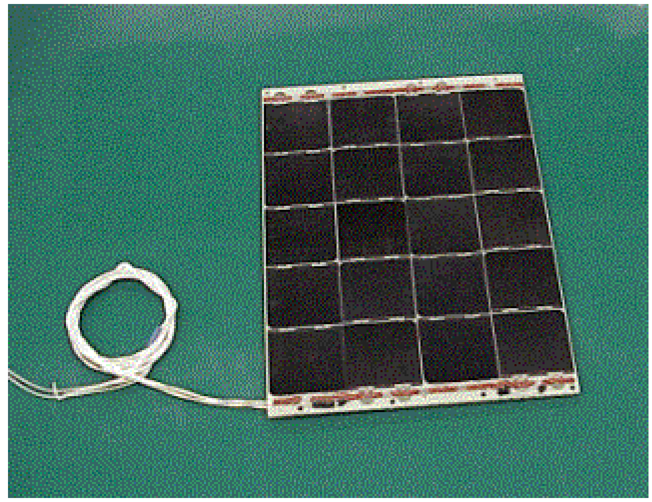



Figure 3: FS-2 solar panels from two sources.
A USAFA-built solar panel is shown on the left. A COTS panel from Spacequest is shown on the right.

The FS-2 battery contains 7 Sanyo N4000DRL NiCAD cells. The cells are housed in a milled aluminum box approximately 6 x 4 x 1 inches. The battery was supplied by Surrey Satellite Technology, Ltd. (SSTL). The battery box design and assembly are in accordance with JSC 20793 *Manned Space Vehicle Battery Safety Handbook*. The cell vent is free of obstruction so that the cell is able to vent excessive pressure in the event of an anomaly. For this reason, the ends of the cell were not potted or covered. The battery uses the following design criteria:

- The battery has a slow-blow polyswitch (3 Amp) incorporated into the electrical circuit, into the negative lead of the battery. See Section 10 of the Battery Acceptance Report in Appendix E of the FSDP for details on the slow-blow polyswitch.
- The battery housing completely encloses the cells.
- There are two venting holes on each of the two lids of the battery box. The lids containing the venting holes are oriented along the spacecraft y-axis. The venting holes are covered with PTFE/GORE-TEX disks, allowing air to escape, but containing any electrolyte leakage.
- Pigmat MAT 301 material is used to contain the electrolyte of all the cells.
- The Battery box is leak tight. The use of a Sigaflex graphite gasket between the case and lids provides a leak tight seal, but maintains electrical continuity.
- The internal surfaces of the battery box are non-conductive and resistant to the electrolyte. The external surface of the aluminum battery box is treated with Alodine 1200 and the internal coating is EccoCoat EP3.

Figure 4 shows an external and internal view of the battery box configuration. Figure 5 shows the alignment of the battery box within the spacecraft.

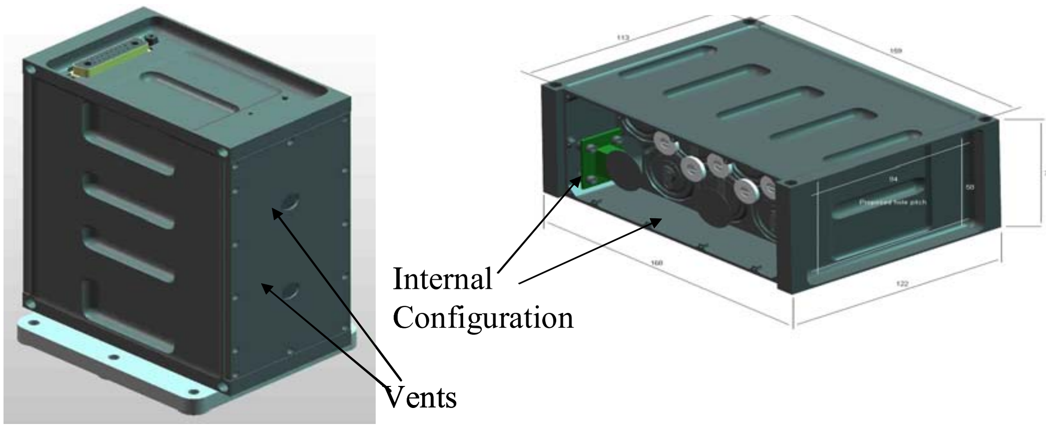


Figure 4: External and internal views of the FS-2 battery box.
(battery vent holes are aligned along the spacecraft y-axis)

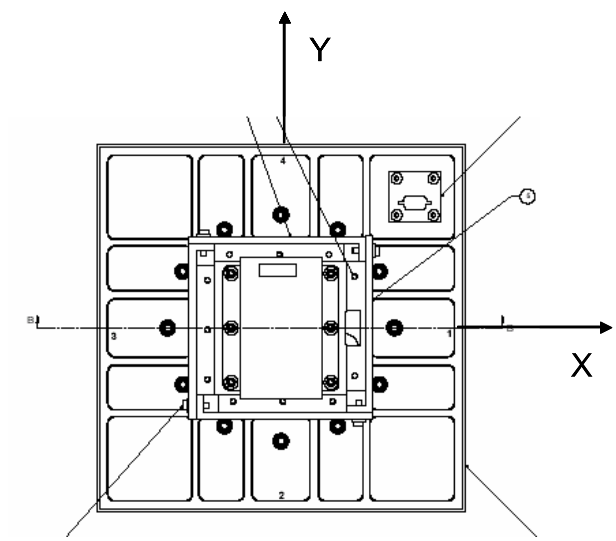


Figure 5: Battery box alignment.

Battery charging, power conditioning and power distribution functions are handled by two sets of commercial electronics using surface-mount components on PCBs. There is a separate Battery Charge Regulator (BCR) for each of the four solar panels. The BCRs regulate the voltage and current from the panels to achieve maximum power point and to the battery to prevent over-charging. The Power Conditioning Module (PCM) is basically a DC/DC converter producing a regulated 5V bus voltage line. The Power Distribution Module (PDM) consists of a set of power switches that can be software controlled to turn other spacecraft subsystems on or off.

All spacecraft wiring and polyswitches are rated to meet NASA Shuttle Payload Bay rating requirements (TA-92-038). To prevent premature (prior to release from the PES) battery charging and application of battery power to any system, a two-fault tolerant inhibit scheme is required. As FS-2 will make use of the un-powered bus exception, four independent micro-switches are used to inhibit all battery operations prior to deployment. The function of each independent inhibit will be independently verified on the ground prior to launch. Two inhibits will be on the positive side of the battery, and two on the ground side. The configuration of these switches will be verifiable via a single access port as discussed in Section 3.4.5.

In addition, a single separation switch is installed before the PCM. This switch prevents stray current from the solar panels, prior to deployment, from causing a soft start of the PCM controller, which could cause it to hang up during subsequent post-deployment operation. As this switch is not a Shuttle safety item, no external monitoring of this switch position was developed. The functional schematic for the overall inhibit scheme is shown in Figure 6. Note that that the BCRs, PCM and PDM are physically housed together in two aluminum trays.

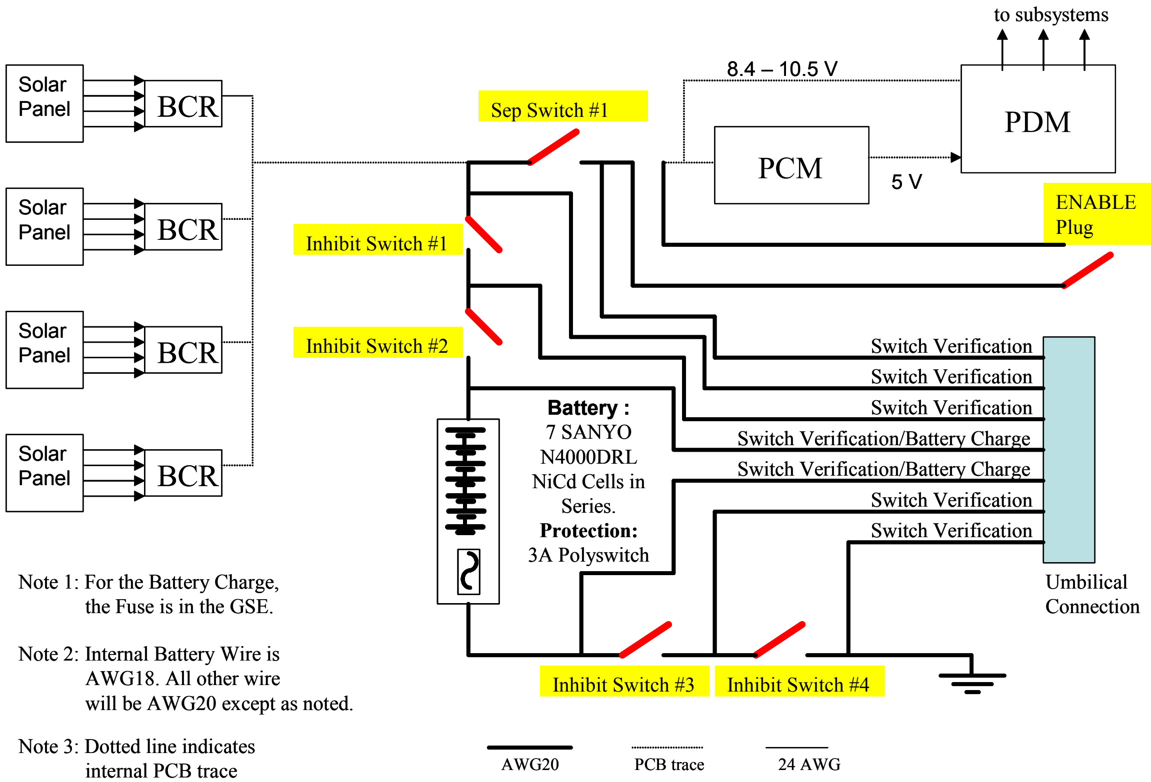


Figure 6: Battery charge inhibits schematic.

The FS-2 Battery Charge Regulator (BCR) and Power Conditioning Module(PCM) consist of COTS Printed Circuit Boards (PCBs) supplied by SSTL, UK. The PCBs are housed in milled aluminum trays, which are mounted to the central column walls of the spacecraft. The BCRs provide hardware-controlled maximum power point tracking of the solar arrays via thermistors mounted on each panel. The PCM provides a regulated 5 V current line and an unregulated line at battery voltage (nominally 8.4 V) and includes switches to turn subsystems on/off. Telemetry and commanding of the BCR and PCM is provided via both a control area network (CAN) node as well as through the onboard computer.

2.3.3 Data Handling Subsystem

The FS-2 spacecraft effectively has two independent data handling systems. The default system relies on the CAN. Each subsystem and the MESA experiment has a unique CAN address. The CAN system offers rudimentary built-in telemetry and commanding ability. In addition, a high speed onboard computer (OBC) based on the StrongArm SA1100 processor provides more sophisticated telemetry and commanding capability as well as dedicated support for specific experiment data collection requirements. MESA experiment interface to both the CAN and the OBC will be through a dedicated System Integration Module (SIM). Both the OBC and SIM were purchased from SSTL as COTS PCBs housed in SSTL-standard 6.5 x 4.8 x 0.82 inch aluminum trays (one tray each for OBC and SIM). Detailed descriptions of key data handling subsystem features are provided below.

2.3.3.1 Control Area Network

The CAN system offers rudimentary built-in telemetry and commanding ability. Each subsystem and the MESA experiment has a unique CAN address (EPS, TX, RX, OBC and SIM). The CAN node controller is a MPC5210 microcontroller with serial links to the OBC. The CAN interface conforms to the CAN 2.0A active and 2.0B passive standards. The bit timing on the CAN bus is 2.6 μs (or 385 kbps). The CAN network is daisy-chained through each subsystem module. Each end of the CAN network is terminated with a 120 Ω resistor.

2.3.3.2 Onboard Computer

The FS-2 OBC is based on the high performance StrongArm processor. This processor was chosen because it combines a high-speed reduced instruction set computer (RISC) with low power (up to 220 million instructions per second (MIPS) @ 190 MHz). The StrongArm processor provides interfaces to static, FLASH, and read-only memories (ROM), power and memory management, integrated clock generation, an on-board real-time clock, an interrupt controller, asynchronous communication ports, and general purpose input/output.

The block diagram for the FS-2 OBC is shown in Figure 7. The OBC has a watchdog timer, 1 Mbyte of flash memory, 4 Mbytes of error detection and correction (EDAC) protected random access memory (RAM), an asynchronous Universal Asynchronous Receiver Transmitter (UART) communication link, an interface to a CAN bus, and a synchronous High-level Data Link Controller (HDLC) link.

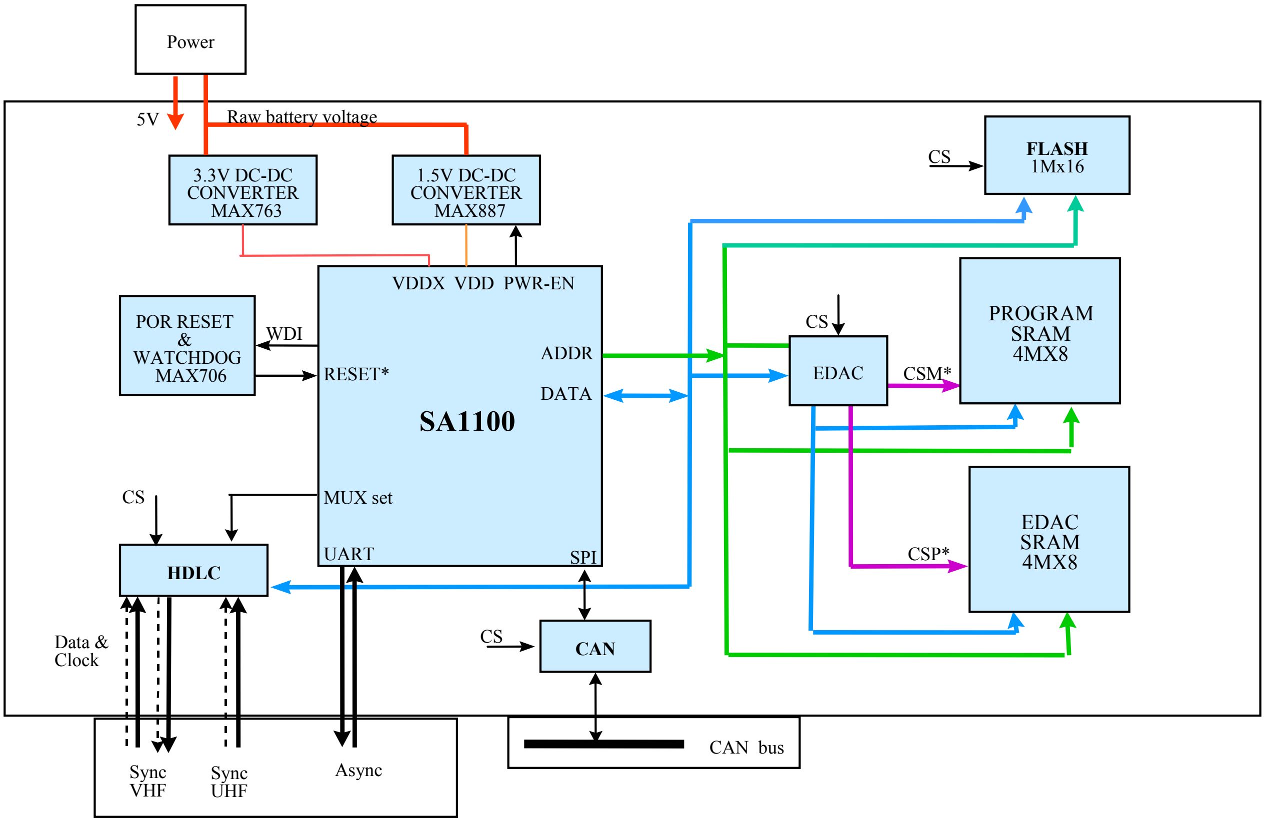


Figure 7: Functional block diagram for the FS-2 onboard computer subsystem.

Key features of the FS-2 OBC include:

- Strong Arm SA1100 Processor
  - High performance RISC processor: up to 220 MIPS @ 190 MHz
  - Low power: <330 mW @ 1.5 V/200 MHz
  - Low power: <230 mW @ 1.5 V/133 MHz
  - Power management: normal, idle, and sleep modes
  - Low voltage: 3.3 V/1.8 V
- 1Mx16 FLASH memory
- 4Mx8 EDAC protected program memory
- WATCHDOG Timer
- CAN bus
- Asynchronous communication link, (uplink at 9.6k baud / downlink at 38.4k baud)
- Synchronous communication link, programmable in a Field Programmable Gate Array (FPGA)
- PCB size: 100 mm x 160 mm, populated on both sides

The FS-2 OBC is designed around the SA1100-EA microprocessor capable of operating at frequencies up to 190 MHz. The CPU clock can be provided by an external clock circuit or by connecting the SA1100 clock inputs with two external crystals of 3.68 MHz and 32.8 kHz. The latter method was chosen. An internal phase-lock loop generates the required internal clock frequency from the 3.68 MHz crystal while the 32.8 kHz crystal is used as a timebase for the real-time clock. To improve stability two 33 pF capacitors connect the 3.68 MHz crystal to ground.

2.3.3.3 Software

Software control of the FS-2 spacecraft via the OBC will accomplished using a single, compiled program written in ASCII standard C. Primary software tasks include:

- MESA sensor control and data gathering
- Orbit propagation for timing of onboard commanded events
- Telemetry collection, real-time downlink and storage
- High-speed automatic data download

Operationally, all mission software will be uplinked as part of spacecraft commissioning. Basic mission software is loaded into the OBC flash memory and will be available on boot-up of the OBC. However, it is anticipated that upgrades to this basic software will be uplinked to improve performance over the lifetime of the mission.

2.3.3.4 Subsystem Integration Module (SIM)

The SIM provides primary data interface between the MESA experiment and the spacecraft bus. The SIM has two connectors, a D-9 for power and CAN bus connections and a D-44 high density for connections to MESA and sun sensors. The signals provided on the payload connector include:

- Eight analog inputs (range 0 - 4.1 V) sampled using a ten bit A/D converter
- Eight analog outputs (range 0 - 4.1 V) generated with eight bit D/A converters
- Sixteen bi-directional digital lines (open drain)
- Asynchronous serial port (TTL signaling)
- Analog reference voltage (4.1 V)
- Power supplies (5 V and Vbatt)

A functional block diagram for the SIM is shown in Figure 8.

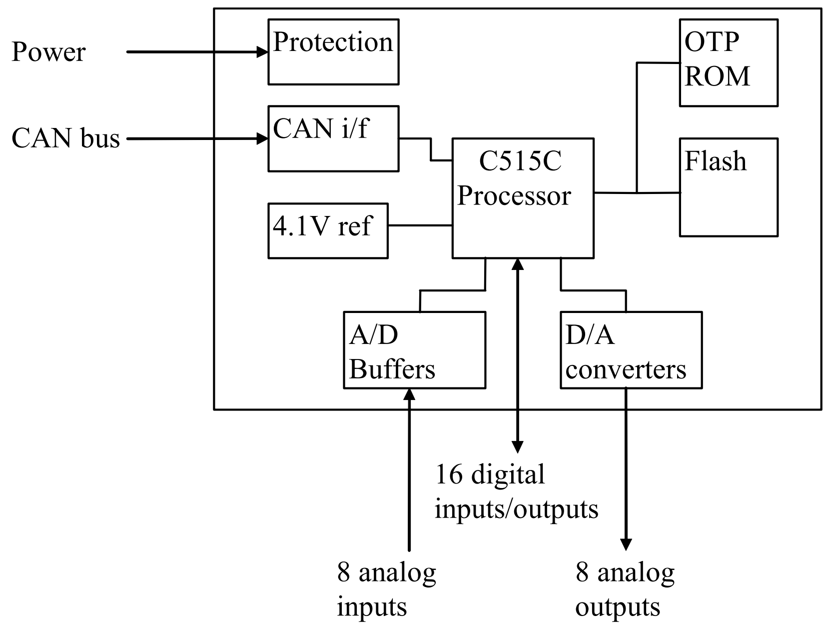


Figure 8: Functional block diagram for the FS-2 SIM.

2.3.4 Communication Subsystems

The FS-2 communication subsystems consist of an S-band transmitter (TX) and a VHF receiver (RX). Both subsystems were purchased from SSTL as COTS PCBs housed in SSTL-standard 6.5 x 4.8 x 0.82 inch aluminum trays (one tray each for the TX and RX). Detailed descriptions of each subsystem are provided below.

2.3.4.1 S-band transmitter

The S-band transmitter will operate at a nominal frequency of 2220.0 MHz at an output power of 500 mW. Power consumption is 4.9 W when the two RF2126 amplifiers are on and 1.5 W when they are powered down. A single S-band ground plane antenna is mounted to the outside top panel of the spacecraft structure and has dimensions of 6.6 inches in height and 2.2 inches in maximum diameter. Figure 9 shows the S-band antenna mounted on FS-2.

The TX employs a CAN controller and interface, and data are sent to the transmitter via the CAN bus or direct from the OBC via the D44 connector. The Downlink modulation scheme uses binary phase shift keying (BPSK) at 38.4k bps at 2.22 GHz. The Transmitter also uses a field programmable gate array (FPGA) to implement the encoders and scrambler, and the 'I' and 'Q' generation.

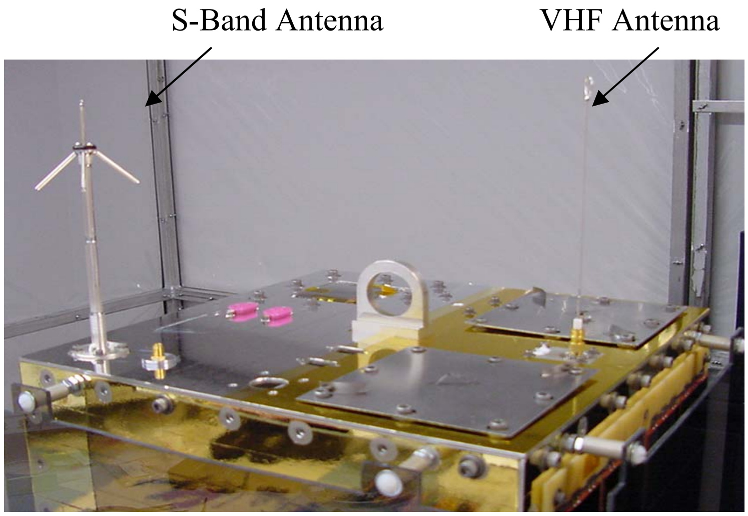


Figure 9: Location of FS-2 VHF and S-band antennas.

A functional block diagram for the S-band TX is shown in Figure 10. Key features of the TX include:

- 38.4k bps @ 2.22 GHz
- BPSK
- 500 mW output power
- CAN Bus interface
- PCB size: 100 mm x 160 mm

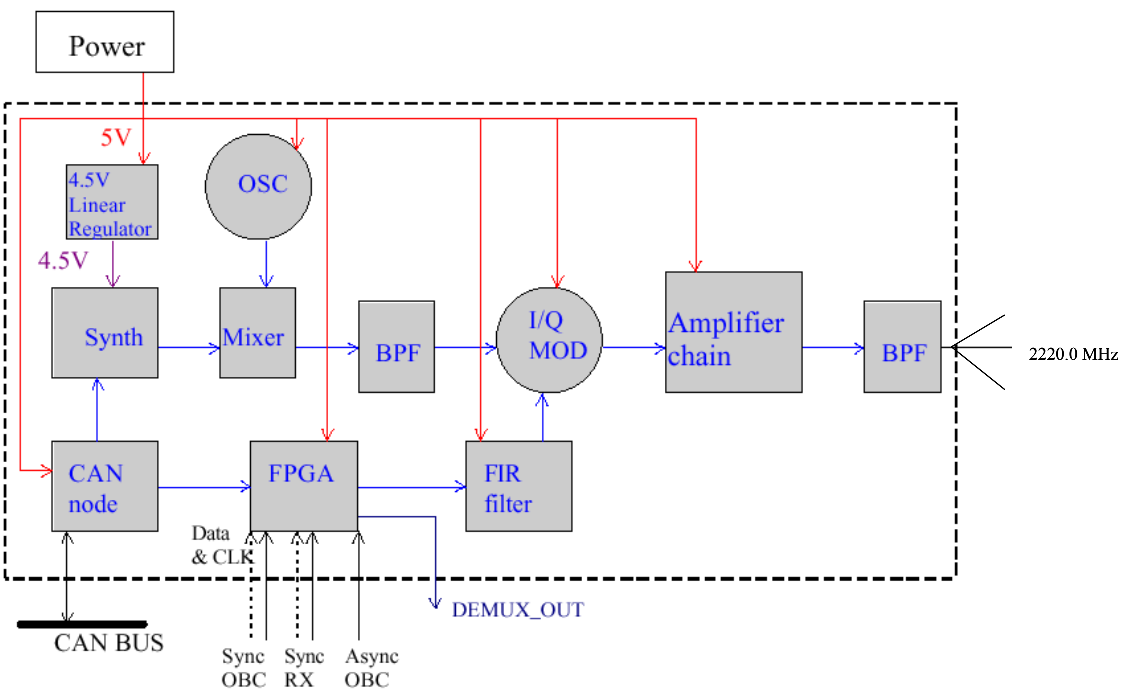


Figure 10: Functional block diagram for the FS-2 S-band transmitter subsystem.

Analysis of the radiated RF power from the S-band transmitter indicates it will be below the payload-to-orbiter limit specified in (ICD-2-19001) with the payload bay doors open.

With the doors closed, four inhibits prevent battery power from being inadvertently applied to the TX. There is no safety concern with the possibility of powering the TX directly from the solar panels. As the analysis described in section 5.2.5 of the Flight Safety Data Package (FSDP) shows, the RF energy is below the limit with PLB doors open; and of course, the solar panels can only generate power in this case.

2.3.4.2 VHF Receiver

A regulated 5 V bus power line powers the VHF Receiver module, and the RX section itself runs off an internally regulated 4 V line. The power consumption is approximately 400 mW. A VHF monopole whip antenna is used, mounted to the outside top panel of the spacecraft as shown in Figure 9 extending 6.9 inches above the spacecraft surface. The front end of the RX consists of a low-pass filter (LPF), a 20 MHz band-pass filter (BPF) and a low-noise amplifier (LNA) circuit. The RX employs a CAN controller and interface, and recovered data can be distributed via the CAN bus or direct to the OBC via the D44 connector. The uplink modulation scheme used is 9600 bps frequency shift keying (FSK) at 148.290 MHz.

Key features of the FS-2 RX include:

- 9.6k bps FSK @ 148.290 MHz
- Low power: 400 mW @ 5 V
- 1st intermediate frequency (IF): 21.4 MHz with matched pair of filters
- CAN Bus interface
- Board size: 100 mm x 160 mm
- Voltage buffered receiver signal strength indicator: 70 dB usable range
- Local Oscillator (L.O.) suppression: 40 – 60 dB

A functional block diagram for the FS-2 RX is shown in Figure 11.

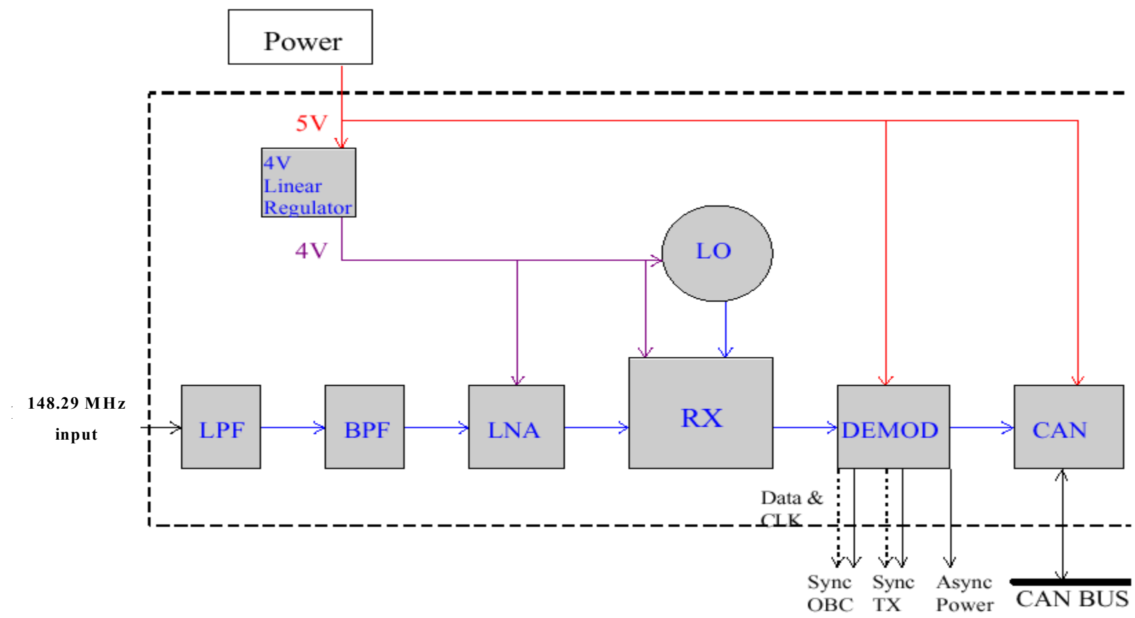


Figure 11: Functional block diagram for the FS-2 VHF receiver.

2.3.5 Thermal Control

The thermal control subsystem's purpose is to maintain the temperature of the spacecraft components within the desired temperature limits specified by the component manufacturer. Additionally, the thermal control subsystem seeks to minimize the thermal cycling, or fluctuation, about the equilibrium temperature of each component. On FalconSAT-2, we are using a passive thermal control subsystem consisting of thermal tapes applied to the exterior of the spacecraft.

Because the baseline thermal behavior of FalconSAT-2 without any thermal control applied is already within temperature limits, our thermal tape design does not need to modify the thermo-optical properties of the structure greatly. However, another consideration in determining the thermal design is the attitude control requirement of inducing a spin to ensure that the solar panels receive sunlight. To do this, thermal tapes with differing absorptivity values will be adhered to each side in a pattern that will induce a spin due to torque created by solar radiation pressure.

FalconSAT-2 thermal control requirements can be summarized as follows:

- Maintain all components within operational limits—The operational temperatures are limited by the electronic components within the satellite, and specifically by the battery. The battery is the most thermally sensitive of the satellite subsystems. As a result, the nominal temperature range targeted for the batteries and internal components of FalconSat-2 is 0 to +30 deg C. The other commercial electronics within the satellite have temperature limits of -40 and +85 deg C. The structural components and solar panels have much more relaxed temperature limits. Table 2 summarizes the temperature limits for FalconSat-2.

- Passive attitude control—Attitude control requirement that must be considered in the thermal design. Differential thermal coatings will be used to impart a spin due to solar radiation pressure torque on the satellite to ensure that the solar panels are always receiving sunlight, and that no one side is constantly pointed at the sun.

After running simulations using the thermo-optical properties of several combinations of thermal tape, we decided to use a combination of aluminum, Kapton, and gold thermal tapes. Specifically, the tapes shown in Table 4 will be used.

Table 4: FalconSAT-2 Thermal Tapes

TAPE	NAME	ABSORPTIVITY	EMMISSIVITY
Aluminum	Sheldahl Second Surface Aluminum Polyimide Tape with 966 Acrylic Adhesive (Item # 146520)	0.14	0.05
Kapton	Sheldahl First Surface Aluminized Polyimide Tape with 966 Acrylic Adhesive (Item # 146385)	0.39	0.63
Gold	Sheldahl First Surface Gold Coated Polyimide tape with 966 Acrylic Adhesive (Item # 146482	0.20	0.02

The thermal tapes were placed on the satellite in a configuration that will induce a torque via solar radiation pressure on the top and bottom of the spacecraft. On the +Z and –Z facets, which are the facets without any solar panels, half of the surface was covered with aluminum tape and half with Kapton tape. This is because our primary objective on these sides is to induce a torque from solar radiation pressure to avoid a situation with no solar panels illuminated by the sun. For the +X, -X, +Y, and –Y facets, which are the facets mounted with solar panels, only gold tape was used. This is because modeling and simulation shows that the satellite tends to run cold, therefore, the maximum a/e ratio tape should be on most sides, while maintaining the solar radiation pressure torque on the +/- Z facets of the satellite. A diagram of the thermal tape design for each facet of the satellite is shown in Figure 12.

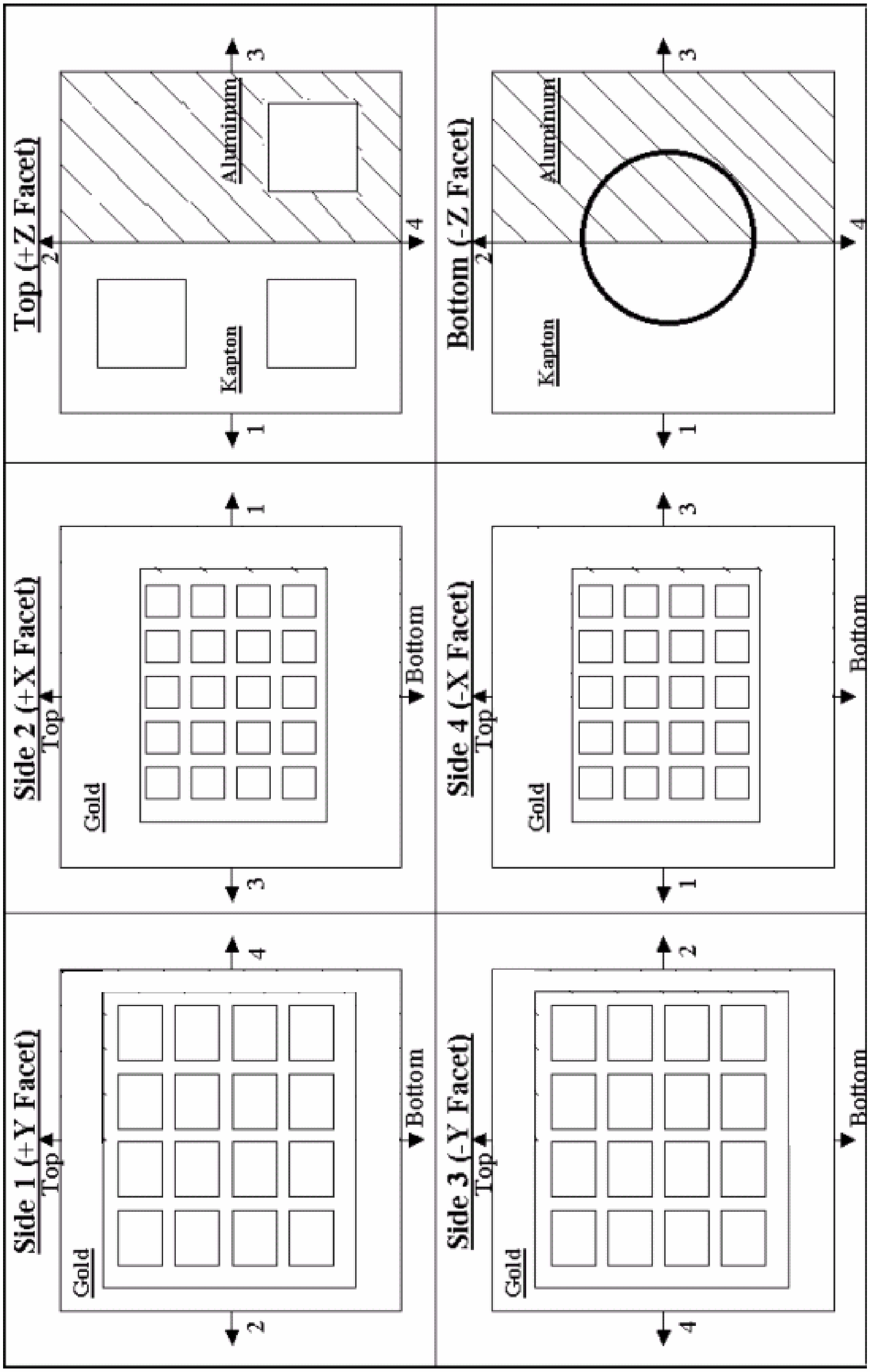


Figure 12: FalconSAT-2 Thermal Tape configuration.

2.3.6 Attitude Determination & Control (ADCS)

FS-2 will use a combination of Sun sensors and solar panel current to determine spacecraft attitude. One Sun sensor is mounted on the top panel of the spacecraft on the same face as the antenna pointed in the spacecraft +Z body direction. A second Earth/Sun sensor is mounted on the base plate pointed in the –Z body direction. Figure 13 shows the FS-2 body axis coordinate system. The Sun sensor is not shown. Each sensor will use a single photo-resistor with an effective field of view of approximately 100°. Spacecraft attitude will be determined post hoc using ground processing of telemetry.

Passive attitude control will be achieved through using atmospheric drag and solar-pressure spin tapes. The spin tapes will provide both passive thermal control, as discussed in section 2.3.5, and provide areas of high or low solar absorptivity. This differential absorptivity is predicted to be sufficient for solar pressure to induce a slight spin. This passive control scheme is designed to ensure the spacecraft does not become inertially locked in an unfavorable attitude such that the solar panels are not pointed at the sun for at least part of each orbit.

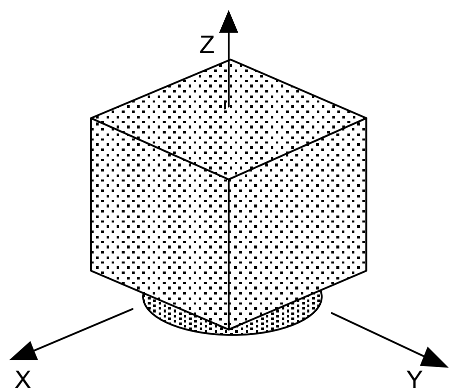


Figure 13: FalconSAT-2 body axis coordinate system.
+Z is out the top plate of the spacecraft. –Z is out through the interface ring.

2.3.7 Miniature Electrostatic Analyzer (MESA)

The primary science objective of the FS-2 mission is operation of the MESA experiment. MESA will not be activated until after FS-2 is deployed from the Shuttle and complete spacecraft checkout and commissioning. It is inhibited from doing so by the 4 inhibits placed around the FS-2 battery as described in Section 2.3.2. An overview of the MESA instrument is presented here to highlight that there are no Shuttle safety issues associated with the presence of MESA. A discussion of MESA science objectives and operations is included here for information only.

The MESA experiment is designed to investigate low latitude ionospheric plasma depletions with a novel, miniaturized electrostatic analyzer. The MESA experiment is an electrostatic analyzer in the form of a patch sensor designed to measure electron fluxes of energies from thermal up to 100 eV. The MESA experiment consists of two laminated analyzers (LA) and a single electron retarding potential analyzer (RPA) that will operate as a suite to provide *in situ* sampling of ionospheric electron density and temperature along the FS-2 orbit track. The MESA experiment will sample the electron density over 6 energy channels and at a rate of 10 spectra per second. The two LAs and the RPA are mounted to the top panel of the spacecraft as shown in Figure 14.

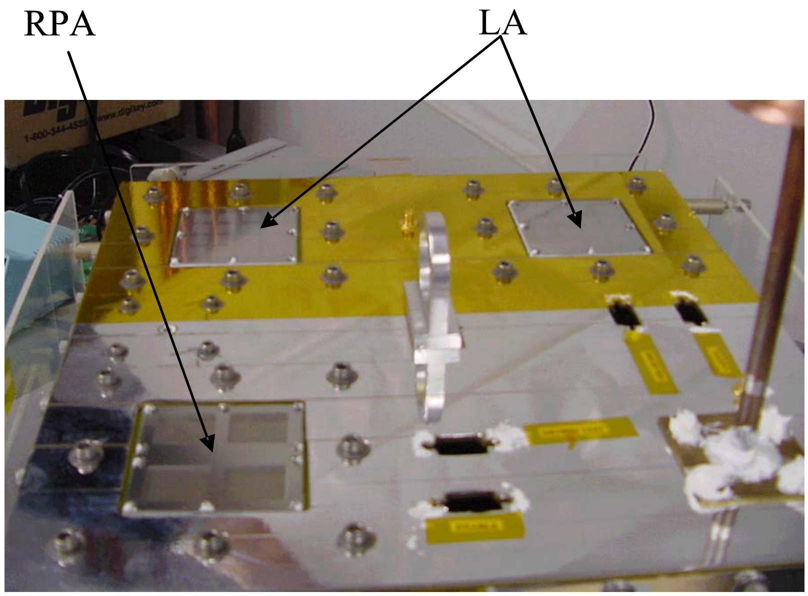


Figure 14: Arrangement of the MESA patch sensors on the top panel of FS-2.

2.3.7.1 Engineering Layout

Both the two LAs and the RPA have the same external dimensions: 3.2 in. x 3.2 in. x 0.40 in. Refer to Figure 15 for the top and side views of the LA. Each sensor consists of a single printed circuit board (PCB) with surface-mount electrical components. A stack of thin stainless steel perforated plates separated by alumina spacers are then attached to the PCB to act as “lenses” to separate out electrons of various energies. The MESA sensors will be mounted such that the stainless steel plate is flush with the spacecraft top surface leaving the PCB inside the spacecraft. A configuration diagram is shown in Figure 16. Mounting to the structure is via 4 fasteners through a bracket on the inside of the spacecraft. Thus, the only exposed portions of MESA are the stainless steel plates. Therefore, there is no risk of shattering the MESA sensors due to kick loads.

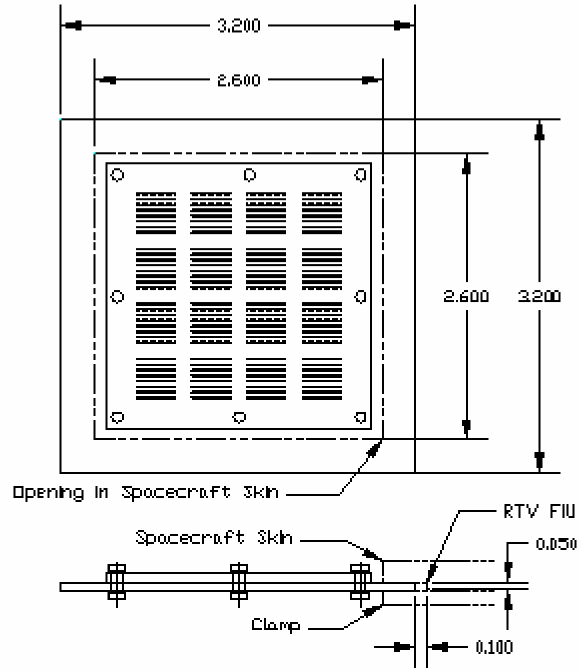


Figure 15: MESA mechanical drawings, top and side views.

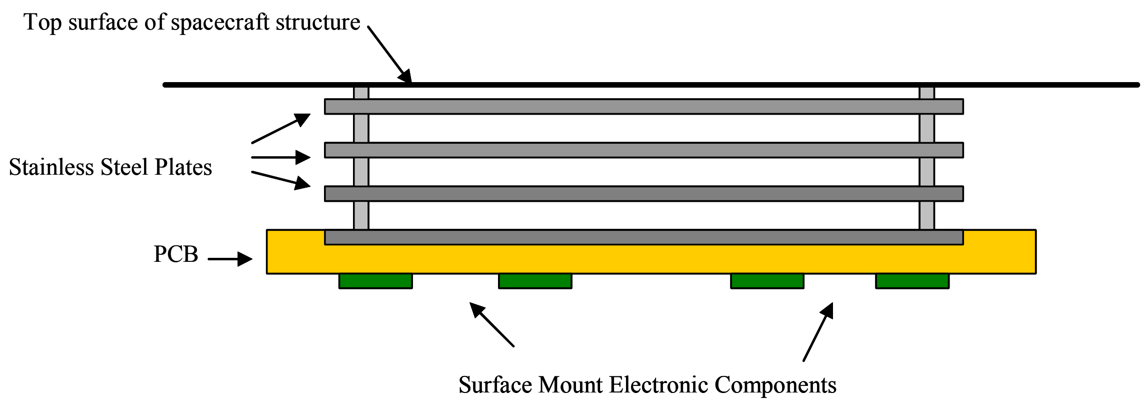


Figure 16: Physical layout of MESA sensor

2.3.7.2 Scientific Objectives

The primary (minimum) objectives of the MESA experiment include: (1) To investigate the morphology of plasma depletions in the F region ionosphere, especially in the low and mid latitudes, and (2) to demonstrate the utility of MESA in the measurement of thermal ionospheric electrons.

The secondary (desired) objective is to investigate the structure and evolution of ionospheric plasma bubbles by taking advantage of opportunities to make *in situ* multi-point measurements of electron densities within a single structure. To accomplish this objective, the MESA experimenters will coordinate a data exchange with experimenters from other Low Earth Orbit (LEO) missions making *in situ* plasma measurements in the ionosphere.

Software control of the MESA sensors is initiated through the Subsystem Integration Module (SIM) and a dedicated MESA Interface Board (MIB). Together, these two PCBs manage all data interface to/from the sensors and OBC. No MESA operations will be initiated until the spacecraft is fully commissioned following deployment.

2.3.7.3 Operations Concept

The primary operating mode of the MESA sensors is the data collection mode during spacecraft eclipse. The MESA data collection will begin 50 km (+/- 35 km) before crossing from the dayside into the penumbra of the Earth. Collection should continue until the spacecraft is exposed to sunlight (*i.e.*, when the spacecraft exits the Earth's penumbra heading into the dayside ionosphere). MESA requires that an average of 15 minutes of eclipse time per orbit should be spent collecting data at magnetic latitudes below 45°.

In the fast data collection mode, the MESA sensors collect 10 spectra of electron fluxes per second. With a 16 bit data word, a 6 channel data collection mode would result in 16 bits × 6 energy channels × 10 spectra per second = 960 bits per second. Since the MESA experiment collects data during eclipse, for an average eclipse time of 45 minutes per orbit, the nominal data collection rate per orbit is 2.59 Mbits/orbit. Data reduction algorithms will be employed to reduce the downlink requirement to a nominal value of 200 kbits/orbit, not to exceed (NTE) 256 kbits/orbit. A slow data collection mode (1 spectrum per second) will be available for contingency purposes.

2.3.7.4 Experiment Success Criteria

The MESA experiment success criteria has been established for each experiment objective (refer to sub-section 2.3.7 introduction). They are as follows:

- Primary Objective 1: The level of success is the fraction of the 6 month minimum experiment duration during which data in fast mode (*i.e.*, 10 spectra per second) were obtained, reduced, and successfully downlinked. For example, if MESA gathered data in fast mode over 4 months only, then the mission would be 67% successful in attaining this particular objective. If the instrument had to be placed in slow mode (*i.e.*, 1 spectra per second,) for greater than 50% of the nominal 6-month mission duration of data collection, then the success rate should be halved.
- Primary Objective 2: The level of success is a) 100% if it is demonstrated that MESA is capable of measuring thermal ionospheric electrons over 6 energy channels at a rate of 10 spectra per second, b) 75% if it is demonstrated that MESA is not capable of making these measurements at 10 spectra per second, but it is demonstrated that samples of 1 (6-channel) spectra per second are feasible and reliable, c) 75% if it is demonstrated that MESA is capable of measuring bulk electron flux to the instrument at a rate of 10 samples per second, but is not capable of resolving the spectra into energy channels, and d) 50% if it is demonstrated that MESA is capable of measuring bulk electron flux to the instrument at a rate of 1 sample per second, but is not capable of resolving the spectra into energy channels.
- Secondary Objective: Success is 100% if MESA passes through a plasma depletion simultaneously with another satellite, with both spacecraft making *in situ* ionospheric plasma measurements. Success is 100% even if there is only one encounter of this type during the entire mission. Success is less when the two satellites enter the plasma depletion at different times: Success = $[1-(\Delta t)/t_{orbit}] * 100\%$, where Δt is the difference between the time the first satellite exits the

plasma depletion and the time the second satellite enters the same plasma depletion, and t_{orbit} is the average orbital period of the two satellites. Since MESA has no control over the orbit of the second satellite, it would be beneficial to select a spacecraft precession rate that maximizes the number of conjunctions with a second, well known ionospheric plasma diagnostic spacecraft (*e.g.,* Air Force Research Laboratory's Communication/Navigation Outage Forecasting System (C/NOFS)) during eclipse and near the magnetic equator.

Minimum MESA experiment success is attained if the average success level of the two primary objectives is greater than 75%.

Desired mission success is attained when the following two conditions are met: 1) the average success level of the two primary objectives is greater than 90%, and 2) the secondary objective success level is greater than 50%.

2.3.7.5 Experiment data downlink

The MESA experiment has neither its own data storage capacity, nor its own transmitter. The experiment will generate approximately 256 kbits/orbit with the application of data reduction algorithms employed by the FS-2 flight computer.

There are no hard requirements on the download times. MESA data will be time-stamped with spacecraft Mission Elapsed Time (MET). Spacecraft latitude, longitude, magnetic local time, and Universal Time, all as a function of MET, will be used for post-processing.

2.3.7.6 Other Environmental Requirements

The MESA operating and storage temperatures are specified in Table 2. MESA has no requirements regarding thermal isolation, humidity, or atmospheric pressure. There are no safety concerns with exceeding these temperatures, however proper functionality of the MESA experiment will not be ensured if exceeded.

2.4 Operations

Deployment and operation of FS-2 hardware occurs in five phases as shown in as shown in Table 5.

Table 5: FS-2 Operational Timeline

Phase	Event	Mission Time	Power Status
0	Awaiting deployment	<T0	Not powered
1	FS-2 ejection from PES, release of micro-switches removes electrical inhibits to EPS, receiver turned on (if sufficient solar and/or battery power available)	T0	Powered
2	Battery charging	T0 to T+10 hrs	Powered
3	First contact, transmitter turned on by ground command	First pass over USAFA >T+10 hrs	Powered
4	Begin commissioning	Subsequent passes over USAFA	Powered

At T0, ejection of FS-2 from the PES, all the micro-switches that inhibit the EPS will close. This will allow power to flow from the solar panels to the battery and the PCM. The mission operations concept calls for a minimum of 10 hours to elapse (approximately 6 orbits) to allow full charging of the batteries before first contact with the satellite is initiated from the USAFA ground station. With power available onboard, only the EPS and RX will automatically be powered on. During first contact, the TX will be commanded on by ground controllers. Following initial contact, spacecraft commissioning will begin. No NASA operational assistance is required after release.

2.4.1 Critical Procedures

No on-orbit critical procedures have been identified for FS-2.

3. PAYLOAD REQUIREMENTS FOR STANDARD CARRIER SERVICES

3.1 Carrier to Payload Electrical Interfaces

FS-2 will require no carrier to payload electrical interface connections.

3.2 Carrier to Payload Mechanical Interfaces

FS-2 is attached to the Shuttle Hitchhiker Pallet Ejection System (PES) via an adapter that is mounted to the bottom surface of the spacecraft. The FS-2/PES will be installed in a canister with lid. The entire assembly is installed in the Shuttle Orbiter Payload Bay. The FS-2 mechanical interfaces meet the standard interface specifications for PES.

3.3 Carrier to Payload Thermal Interfaces

The payload meets the standard thermal interface requirements specified in the CARS for PES payload in a canister with HMDA and heaters. The FS-2 thermal design will rely exclusively on passive thermal control consisting of judiciously selected surface coatings post-deployment.

3.4 Ground Operations Requirements

This section outlines the plan for FS-2 ground operations at GSFC and the Kennedy Space Center (KSC). Ground operations consist of physical examinations, functional & verification testing, servicing, and final flight preparation, including removal of Remove Before Flight (RBF) covers, see Table 6. It is requested that all activities be accommodated at both GSFC and KSC, except for final flight preparations, which are conducted only at KSC. Areas at KSC at which the FS-2 may be accessed include the Payload Processing Facility (PPF), the Orbiter Processing Facility (OPF), and the launch pad (PAD).

Table 6: Ground Operations Locations

Ground Operation	GSFC	KSC	Note	Ref Section
Physical Examination	YES		Inspection of fastener torques and integrity of separation systems	3.4.1
Pre-PES Installation	YES		Removal of RBF covers, Insert and secure ENABLE plug, Battery charging	3.4.2
Functional Testing	YES	PPF	RF transmission and reception	3.4.3
Verification Testing	YES	PPF	Safety inhibit verification	3.4.3
Servicing	YES	PPF	Remove solar panel covers, Battery top-charging, Battery conditioning	3.4.4
Final Flight Preparations		PPF, OPF	Battery top-charging, removal of RBF covers	3.4.5
On-PAD Battery charging		PAD	Battery top-charging	

Note for Table 6.

1. Only PAD requirement is for battery charging on a non-interference basis only if time since final PPF top-off charge exceeds 45 days (see Section 3.4.4).

3.4.1 Physical Inspection (GSFC/KSC)

At GSFC site, FS-2 will be subjected to a post delivery inspection to determine the payload's physical condition following transport, handling, and the passage of time. Physical examination involves checking the exterior surface for scratches, dents, and sharp edges; breakage of solar cells; fastener torques; and separation systems mechanical integrity.

3.4.2 Pre-PES Installation

Prior to installation of FS-2 on to PES, several operations will need to be performed:

- Top charging of the battery (depending on elapse time since previous charging)
- Removal of all RBF covers
- Insertion and securing of the ENABLE plug

Three RBF sensor covers and four RBF solar panel covers must be removed as shown in Figure 17. A summary of all RBF and IBF items is detailed in Table 7.

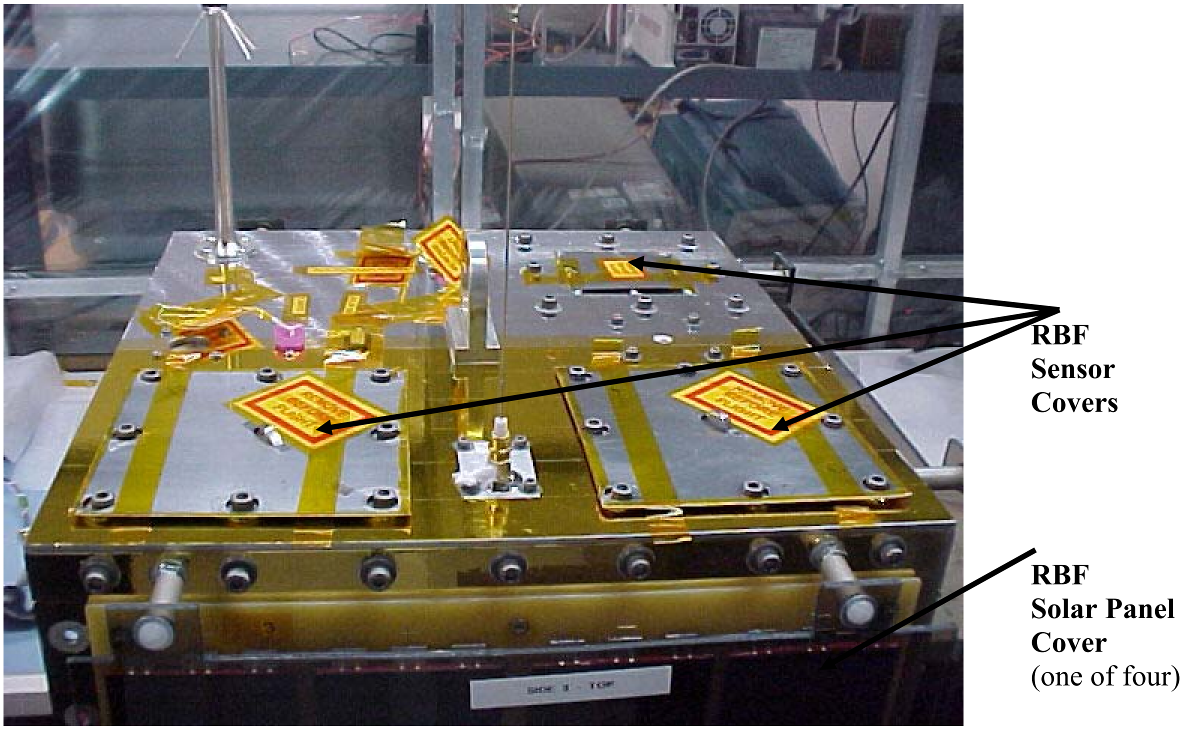


Figure 17. Locations of RBF/IBF Covers

Table 7: FS-2 Locations of RBF/IBF Items

Name	Type	Physical Location	Description	When removed/installed
Solar Panel Covers	RBF	Over each solar panel (4 total) (see Figure 17)	Plexiglass covers secured to the side of the spacecraft with fiberglass fasteners during transport and handling	Prior to installation in the PES container
MESA/RPA Covers	RBF	Over each MESA/RPA sensor (3 total) (see Figure 17)	Aluminum covers secured to the top of each sensor with kapton tape during transport and handling	Prior to final HH can closeout
ENABLE plug	IBF	Top Panel (see Figure 18)	Female D9 connector with pin-to-pin wiring to make appropriate connections within the spacecraft circuitry	Prior to installion of the spacecraft into the PES
CAN/TTL Port Cover	IBF	Top Panel (see Figure 18)	Plastic D9 connector cover	Prior to final spacecraft closeout

			secured with RTV	
Solar Simulator Power Port Cover	IBF	Top Panel (see Figure 18)	Plastic D9 connector cover secured with RTV	Prior to final spacecraft closeout
Switch Test/Battery Charging Port Cover	IBF	Top Panel (see Figure 18)	Plastic D9 connector cover secured with RTV	Prior to final spacecraft closeout (may have to be removed and re-installed to support battery charging on the PAD as necessary)

The ENABLE plug must be installed during pre-PES installation to allow FS-2 circuitry to operate as intended. The ENABLE plug acts as an additional spacecraft functional inhibit for ground testing and handling purposes. The ENABLE plug must be installed before flight (IBF) into the ENABLE port, which is a male DB-9 connector on the top panel of the satellite as shown in Figure 18. The ENABLE plug consists of a female DB-9 connector containing the enable wiring as shown in Figure 19. This plug simply shorts the appropriate pins to allow the spacecraft to function. There are small hold-down screws as part of the ENABLE plug which will be tightened and secured with RTV for flight.

The status of all inhibits will be verified as necessary through the Switch Test and Battery Charge port, which is a female DB-9 connector as shown in Figure 18. This verification process utilizes a temporary connection.

3.4.3 Functional and Verification Tests (GSFC/KSC)

Functional and verification tests will be required to identify any FS-2 functions that may have been affected by transport, handling, or the passage of time.

3.4.3.1 Functional Testing

FS-2 functional testing will include supplying external bus power to activate all electronic components, after which each subsystem will be methodically tested according to a documented set of procedures (note: external bus power will be supplied through ports as shown in Figure 18. These functional tests will require a power supply, laptop computers, and portable GSE. FS-2 will furnish this equipment as well as any specialized cables and wiring.

3.4.3.2 Verification Testing.

Verification testing includes testing of safety inhibits to verify their status. Verification testing will be conducted through a port at the top surface of the payload as shown in Figure 18. The FS-2 program shall provide equipment and procedures for verification testing. Any inhibits removed during testing will later be enabled as part of the test procedure and later re-verified.

In addition to verification testing of safety inhibits, EMI/EMC testing will be conducted at GSFC (by NASA personnel) to test effects of radiated emissions of the type expected in the Orbiter Payload Bay (PLB). It is requested that functional and verification tests on FS-2 be allowed to take place before and after EMI/EMC testing. This will permit the FS-2 program to determine the source of any anomalies.

3.4.4 Servicing Operations (GSFC/KSC)

Servicing operations for FS-2 include removal of solar panel covers, solar cell cleaning and battery charging and battery conditioning. Solar cell cleaning will take place after removal of the Plexiglas solar panel covers prior to integration of FS-2 into the PES container.

Maximum FS-2 battery charge characteristics are shown in Table 8. The batteries, once charged, will be acceptable for approximately 45 days. However, battery top charging is desirable whenever the schedule permits, even if the maximum interval between charges has not been reached. Charging/discharging of the payload batteries will be conducted through a Switch Test and Battery Charging port, which is a female DB-9 connector, located at the top of the spacecraft as shown in Figure 18. Equipment for charging the batteries will be provided by the FS-2 team.

Table 8: FS-2 Battery Charging Characteristics

Charge Characteristic	FS-2
Charge current (mA):	1.0 A
Charge voltage (V):	13 V
Charge time (h):	4.0 h

Notes for Table 8:

1. 7x SanyoN4000DRL NiCd cells (series)
2. All batteries may be charged through port as shown in Figure 18.

Battery conditioning will consist of 2-3 discharge/charge cycles. It is anticipated that this procedure will only be needed once after the planned EMI testing at GSFC.

3.4.5 Final Flight Preparation (KSC)

Final flight preparations include ground operations conducted just prior to rollout.

In the OPF, FS-2 battery top charging is requested as late as possible, but at a minimum the battery must be recharged every 45 days. Once the shuttle is at the launch pad, top charging of the FS-2 batteries is required if 45 days have passed since the previous charge. It is requested that access be provided to allow battery top charging both in the OPF and on the PAD should it become necessary. As stated in Section 3.4.4, access to batteries is possible through Switch Test/Battery Charge port on the top of the spacecraft (see Figure 18).

Once proper operation is verified and final battery charging is complete, the three remaining unsealed ports (Switch Test and Battery Charging port, CAN/TTL pot, and Solar Simulator Power port) will be sealed plastic D9 connector covers secured with RTV.

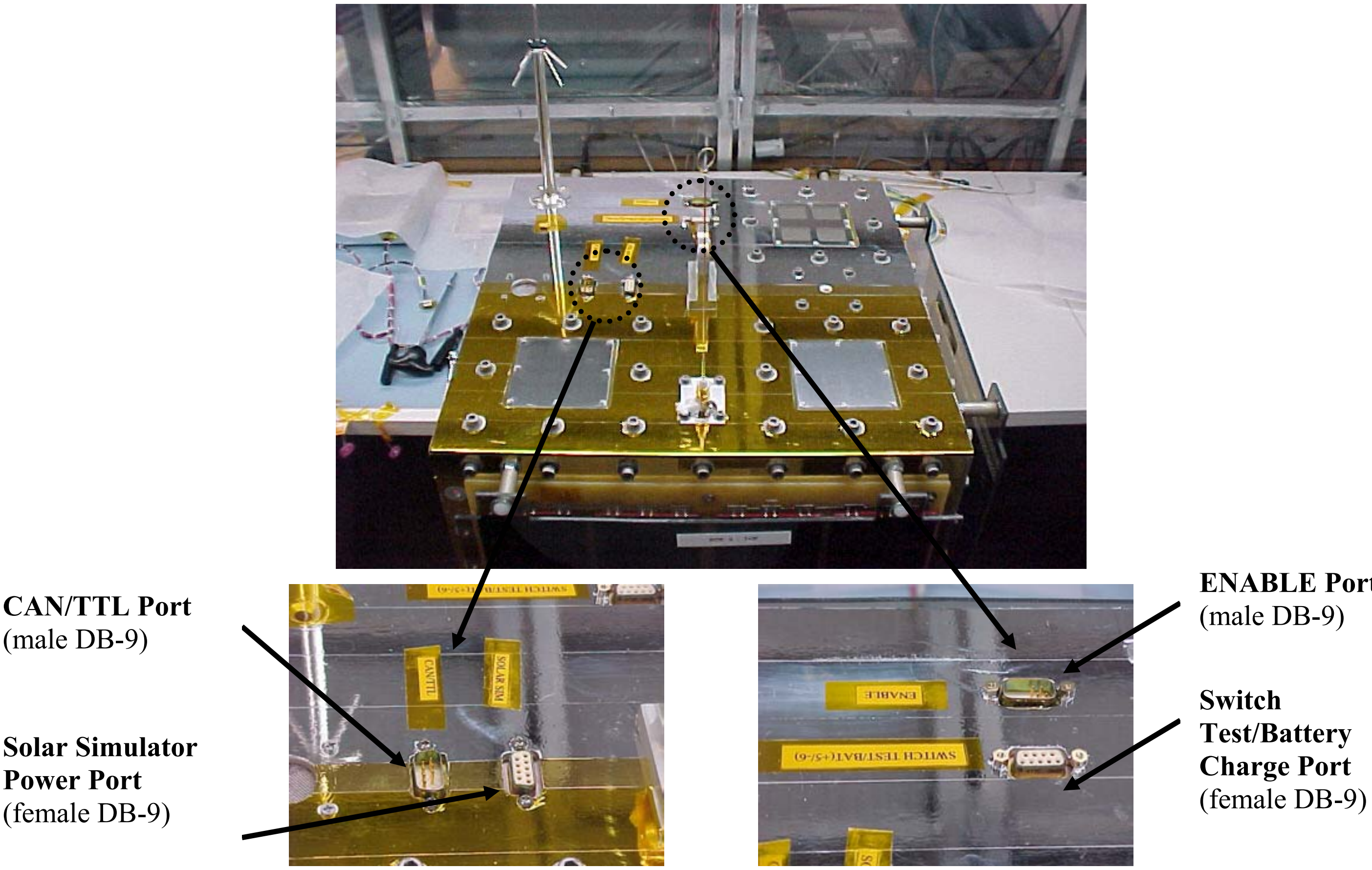


Figure 18: FS-2 Access for Servicing and Functional/Verification Testing

Figure 19. ENABLE Plug

3.4.6 On-PAD Battery Charging

Battery charging on the PAD will only be required if access to the payload bay is available and greater than 45 days have elapsed since the last battery charging. Battery charging equipment and procedures will be the same as described in Section 5.3 and Section 5.4. One additional procedure that will be required is the removal and re-installation of the Switch Test/Battery Charging Port Cover. The procedures for these will be as follows:

1. Attach tether to Switch Test/Battery Charging Port Cover
2. Secure other end of tether to operator
3. Carefully remove RTV securing Switch Test/Battery Charging Port Cover to spacecraft
4. Remove Switch Test/Battery Charging Port Cover
5. Complete battery charging procedures as described in Section 5.4
6. Re-instert Switch Test/Battery Charging Port Cover
7. Secure Switch Test/Battery Charging Port Cover to spacecraft with RTV
8. Remove tether

4. MISSION OPERATIONS REQUIREMENTS

4.1 Operational Scenario

4.1.1 Deployment Time

FS-2 should be deployed as soon as possible into the mission, within 5 to 15 minutes after orbit sunrise. This corresponds to time T0.

4.1.2 Orbit Parameters

It is understood that most orbital parameters will be constrained by the primary mission to service the ISS. For maximum orbit lifetime, deployment should be made at the highest possible altitude.

4.1.3 Orbiter Pointing

The Orbiter payload bay should be oriented such that the sun will fully illuminate at least one FS-2 solar panel at the time of deployment. This translates to an Orbiter Z-axis of 90° $\pm$10° with respect to the sun vector. If possible, deployment should be in a generally posigrade direction.

4.1.4 FS-2 Ejection from PES

Once the Orbiter has achieved the required minimum orbit and pointing attitude, ejection of FS-2 may begin. FS-2 should be ejected at a relative speed of ~3 ft/sec. Ejection should take place no sooner than 5 min after orbital sunrise and no later than 15 min after orbit sunrise. Releasing the system at this time will ensure sunlight is on the solar panels producing the required minimum voltage and to allow the system to be observed and photographed during daylight as well as to permit FS-2 to begin battery charging at >T0 (until then, the FS-2 battery is prevented from charging by the use of four microswitch electrical inhibits).

There are no requirements for deployment over CONUS.

4.2 Experiment Power

Not Applicable

4.3 Experiment Commanding

Not Applicable

4.4 Experiment Telemetry

Not Applicable

4.5 Crew Involvement

Crew involvement is limited to the following:

- Preparation for and initiation of deployment of FS-2 from HH canister.
- Providing video and still images of FS-2 prior to separation from PES. Views of the forward and aft aspects of the payload are requested. This activity is requested in order to document the physical condition of the payload prior to release from the payload bay.
- Providing video and still images of FS-2 during the separation event. Views of the forward and aft aspects of the payload are requested. This activity is required in order to document the physical condition of the payload upon release from the payload bay as well as to document the operation of the PES separation system.

4.6 Instrument Field of View

Not Applicable

4.7 Contamination Constraints

FS-2 is inactive while it is stowed in the Orbiter. It is therefore insensitive to payload bay lights and other electrical activity occurring in the orbiter. EMC characteristics will be confirmed prior to launch. Since experiments will not be conducted until the payload is beyond the influence of the Orbiter, vibration effects due to RCS activity is not considered to be an issue. In addition, the risk of liquid or particulate contamination is considered low, since the payload is partially shielded from the environment via the canister. However, during its deployment, FS-2 will be exposed to physical contamination, and use of the RCS should be limited to avoidance maneuvers. Water dumps and flash evaporator operations should also be avoided during deployment.

4.8 Air/Ground Communications

Request that Air-to-Ground communications be made available to USAFA during deployment to enhance student involvement with the project.

4.9 Customer Supplied Ground Support Equipment

FS-2 CGSE preliminary requirements are summarized in Table 9. Section 5 outlines in detail the specific hardware components that comprise the CGSE. In summary, the major components are three laptop computers, one RF test set, one oscilloscope, one DC power supply, one satellite demodulator, one integrated "Silver Suitcase" which contains the remainder of the RX/TX interface hardware, one Inhibit Switch Verifier, and a programmable battery charger.

Table 9: Customer Ground Support Equipment Characteristics

Item	GSE Characteristic	Applicability
1	Weight	150 lbs
2	Number of assemblies	6-7
3	Power required	115-120VAC
4	Floor space required	10' x 20'
5	CGSE will generate uplink commands	YES
6	CGSE will receive low-rate customer data	YES
7	CGSE will receive medium-rate data	YES

8	CGSE will receive attitude data	NO
9	Number of standard 15a, 115vac, 60-hz outlets	2-3

Note: All items are Payload specific/unique GSE and do not require GSFC HH GSE

4.10 GSFC Payload Operations Control Center (POCC) Requirements

The GSFC POCC will be staffed by FS-2 personnel until deployment. FS-2 personnel will require E-mail/Internet access and long-distance phone access.

4.11 Post Mission Data Products

FS-2 requests the following post mission data products:

- Orbit attitude data (at deployment, LVLH, pitch, roll and yaw); state vector (pre-/post-deployment)
- Ejection film and video
- Audio of deployment timeframe: crew cabin and air-to-ground.

5. GROUND SUPPORT EQUIPMENT

FS-2 personnel will operate three distinct sets of ground support equipment (GSE) at both GSFC and KSC. These are:

1. Functional Testing Equipment
2. Inhibit Switch Verification Equipment
3. Battery Charging and re-conditioning Equipment

This section will describe the hardware, and if applicable, software, for each of these sets of equipment.

5.1 Functional Testing Equipment

As the name implies, the purpose of FS-2 Functional Testing Equipment (FTE) is to provide hardware and software interface to the spacecraft to allow functional verification of subsystems and payload. This section will describe both the FTE hardware and software.

5.1.1 FTE Hardware

A block diagram of the FTE is shown in Figure 20. Individual functional components of the FTE are summarized in Table 10. The numbered items refer to items in Figure 20.

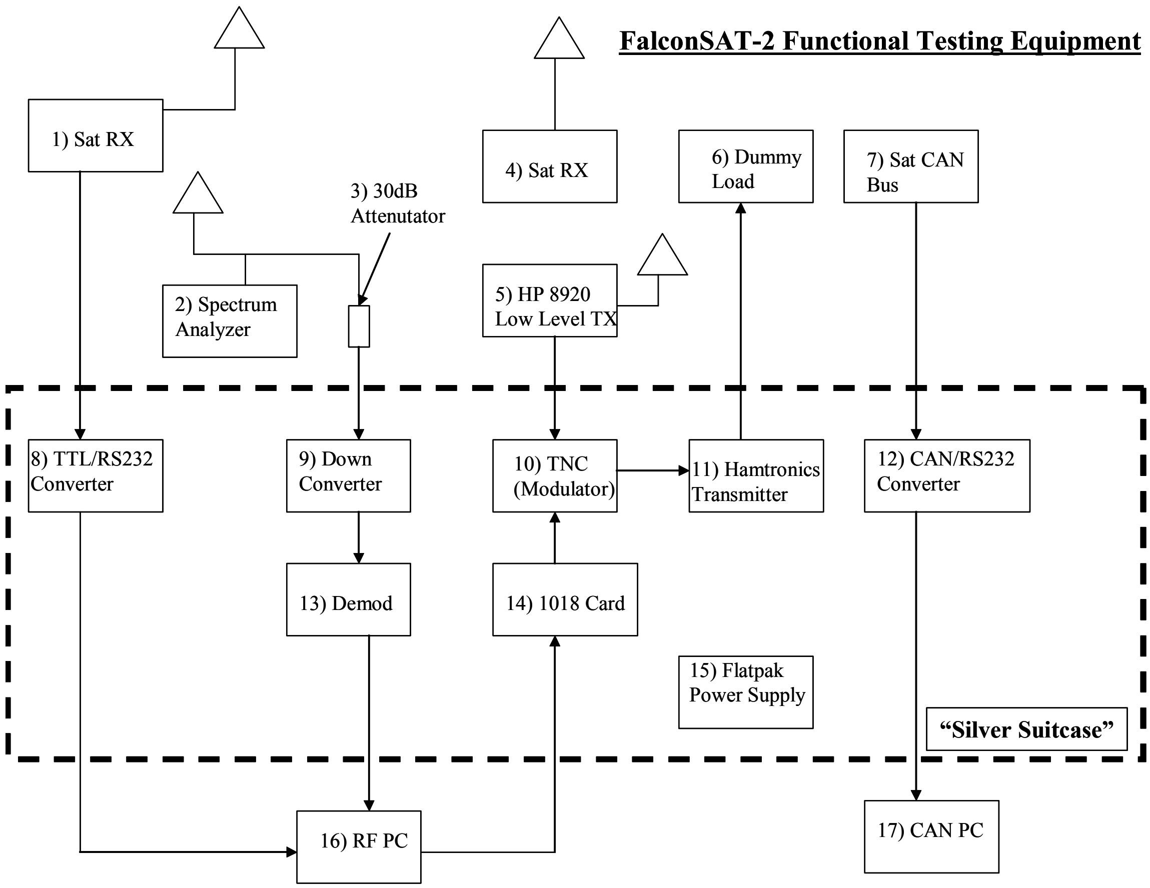


Figure 20: FalconSAT-2 Functional Testing Equipment

The hub of the FTE is a "silver suitcase" that provides modulators, demodulators, power supplies and interface for connectors. A photograph of the "silver suitcase" is shown in Figure 21. In addition, the FTE employs 3 PCs, 2 for telemetry and commanding of the spacecraft and one to control the S-band demodulator. These PCs are shown in Figure 22. Finally, FTE uses both a spectrum analyzer and RF Comm Test set for monitoring and controlling RF communications to/from the spacecraft. These items are shown in Figure 23. A DC power supply referred to as the "solar sim" power supply to provide spacecraft power during testing, which is connected to the Solar Simulator Power Port as shown in Figure 18. An umbilical connection is also made to the satellite to verify the internal functionality through the spacecraft data handling bus, which is via the CAN/TTL Port, also shown in Figure 18.

Table 10: Summary of FS-2 Functional Testing Equipment Components

Item Number (ref. Figure 20 **)**	**Item**	**Description**
1)	Satellite Transmitter	FS-2 S-Band (2.22 GHz) transmitter output
2)	Spectrum Analyzer	Used to analyze the 2.22 GHz signal coming out of the Satellite Transmitter.
3)	30 dB Attenuator	Signal attenuator
4)	Satellite Receiver	FS-2 VHF (148.290 MHz) Receiver input
5)	HP 8920 Low-Level Transmitter	RF Comm test set used as low-level TX
6)	Dummy Load	Attenuator
7)	Satellite CAN Bus Tap/Harness	Data output from spacecraft CAN
8)	TTL/RS 232 Converter	Level convert TTL to RS232
9)	Down Converter	Takes the 2.22 GHz signal from the Satellite Transmitter and converts it to a 70 MHz signal.
10)	TNC (Modulator)	Terminal Node Controller
11)	Hamtronics Transmitter	Low-level TX
12)	CAN/RS 232 Converter	Level converter CAN to RS232
13)	Demodulator	Takes the 70 MHz signal coming out of the Down Converter and removes the modulation scheme so the signal can be interpreted by the software loaded on the RF PC C2.
14)	1018 Card	Asynchronizer
15)	FlatPak Power Supply	Provides power to the internal components of the GSE suitcase. Provides power in the form of 5, 10, & 12 Volts DC.
16)	RF PC	PC running TLM/CMD software to interpret telemetry and send commands over the spacecraft RF link
17)	CAN PC	PC running TLM/CMD software to interpret telemetry and send commands over the spacecraft umbilical link

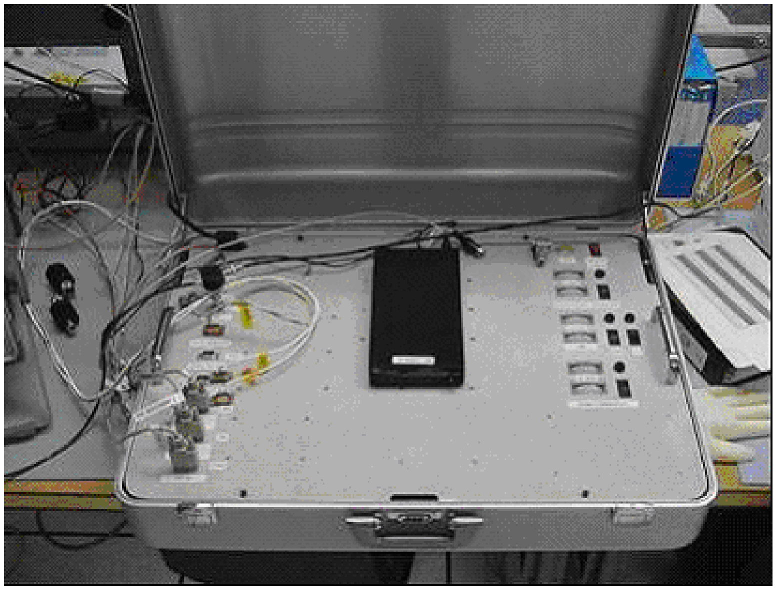


Figure 21: Photograph of the "silver suitcase" used in the FS-2 FTE. This serves as the hub of the FTE housing power supplies, connector interface, and other equipment.

Figure 22: Photograph of FS-2 PCs and demodulator.

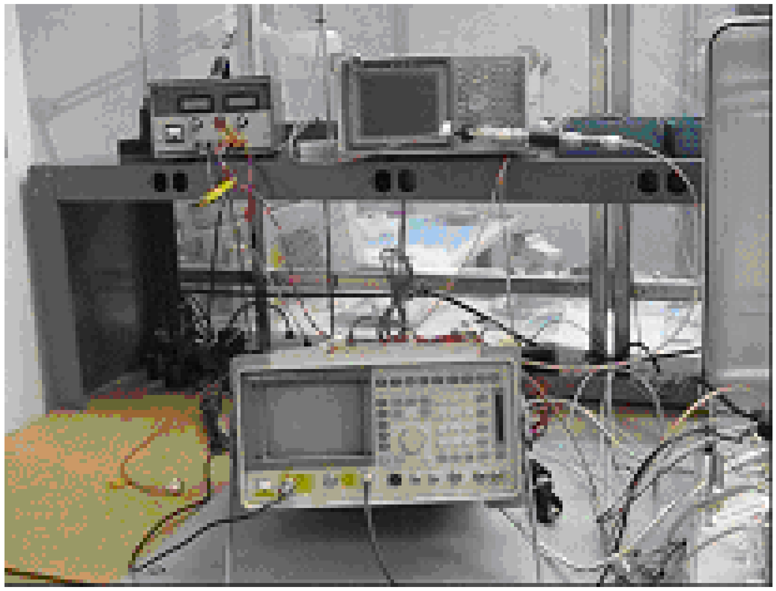


Figure 23: Photograph of FS-2 Spectrum Analyzer, RF Comm Test Set and Solar Sim power supply used in the FTE.

All power for the FS-2 FTE is provided by standard 110 VAC as shown in Figure 24.

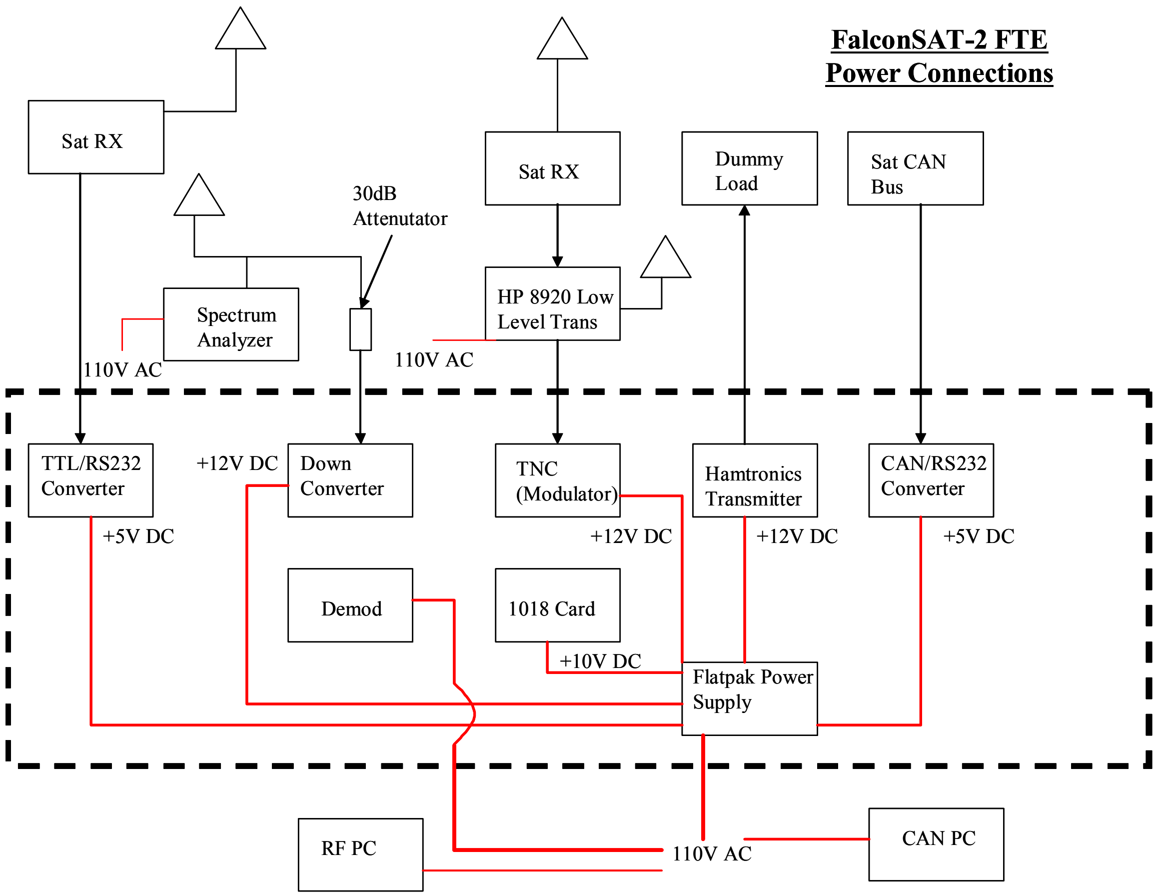


Figure 24: FTE power connections.

5.1.2 *FTE Software*

The primary FS-2 unique software used in the FTE is the CAN/TLM telemetry program provided by Surrey Satellite Technology Limited (SSTL). This software allows operators to monitor telemetry from and send telecommands to the spacecraft using the spacecraft-unique BZL operating system and SNAP bootloader.

Upon startup this dialog-based program presents the user interface shown in Figure 25.

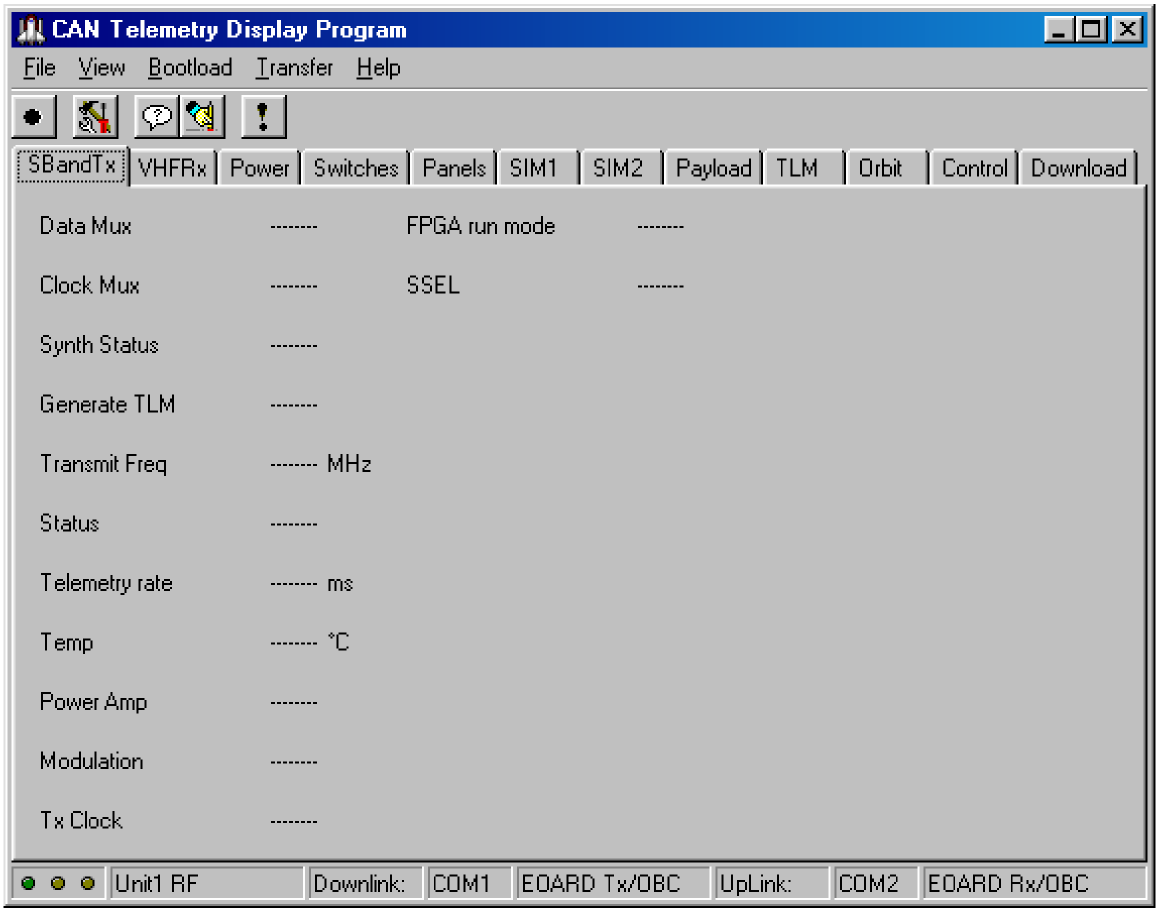


Figure 25: FS-2 FTE software graphical user interface

The default screen shown in Figure 25 indicates a telemetry screen when it is receiving no telemetry. Provided the onboard transmitter is configured to send telemetry back to the ground (as opposed to the on-board computer), and assuming that telemetry requests have been enabled (using the  button), the screen will look as shown in Figure 26.

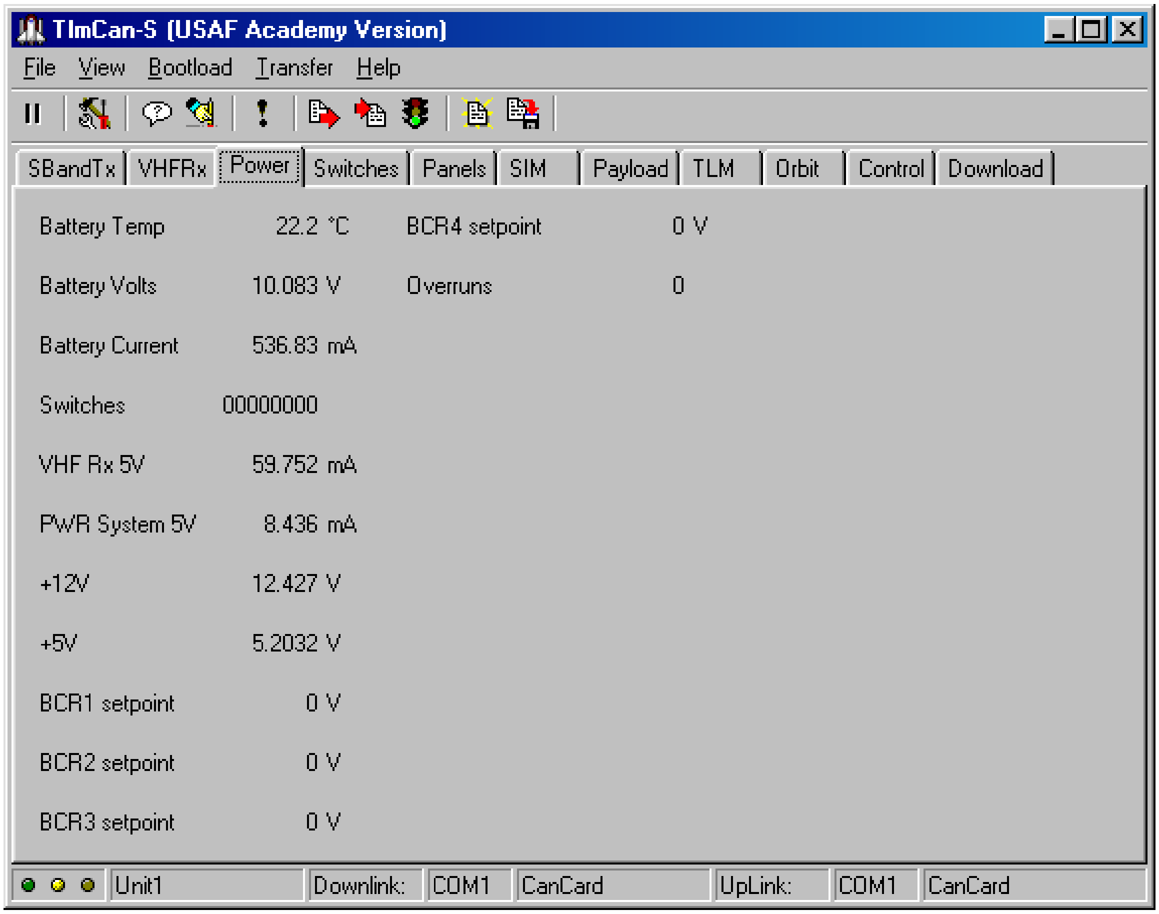
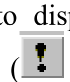


Figure 26: FS-2 FTE software interface showing telemetry data

Telemetry values and calibration curves can be added or deleted by editing a comma-delimited text file (.CSV) file that is easily editable in the Microsoft Excel program. This .CSV file includes labels, telemetry nodes, channels, and values. In addition, it contains calibration curves to allow conversion of telemetry information in counts to their real-world values. The name of the file we will use to specify telemetry for the FalconSat 2 flight model is called FMfalconsat2_tc_Labeled.csv. Details of the file format are given in a later section.

In addition to displaying telemetry, this program can send telecommands, by pressing the telecommand () button. When pressed, a modeless dialog box pops up as shown in Figure 27.

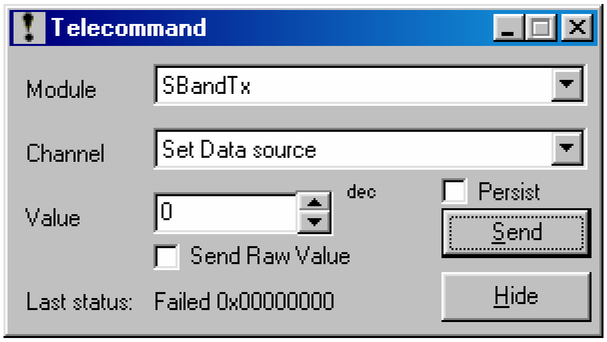


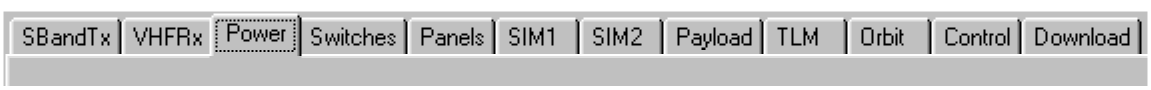
Figure 27: FS-2 FTE software command dialog box

The user may specify the module to which they desire to send a telecommand, the telecommand channel (which usually represents the actual command), as well as a value (typically an ON/OFF value, or a value to set in the device).

As with the displayed telemetry, the telecommands can be specified by the user by editing a separate comma-delimited file (.CSV file). The file we will use for the FalconSat 2 flight model is FMfalconsat2_tc_Labeled.csv. When viewed in Microsoft Excel, a row of this file looks like this:

;Title	Units	Channel	Type	Mask	Min	Max	Low	High	X2	X1	1	High	Low
[SBandTx]													
;Data Mux	-	0x00040015	3	0x00000F00	0	15	-1	-1	0	1	0	-	-

First let's look at the Title column. The first entry, in brackets [], is the name of the module for which telemetry is displayed. This name shows up as the label on the tab on the telemetry page. For example, [SbandTx] is the first module, and can be viewed as the first tab on the telemetry page:



The entries not in brackets are the labels of the individual telemetry channels. The labels also have a "comment" feature. If the first character in the title field is a semi-colon, that channel will not be displayed on the telemetry page. That is what is shown in the "Data Mux" channel entry above.

Why would you want to "comment out" a telemetry channel? Well, you may want to decrease the clutter at one time, but still want the capability to view the telemetry channel in its entirety at a later date. By commenting it out, you can display the channel at a later date by simply deleting the semicolon, as opposed to reentering all of the data.

The units column simply allows you to display the units of the value of that channel on the screen. The 8 digit hex value for the channel actually combines two numbers. The 1st 4 hex digits specify the CAN (Computer Area Network) address of the device that measures the telemetry (i.e. the power module, the transmitter, etc. – each device has its own CAN address). The 2nd 4 hex digits specify the telemetry channel you are requesting from that device. For example, to get the Data MUX switch settings you are requesting the DATA MUX channel (Hex 0015, decimal 21) from the S-Band Transmitter (HEX address 0004, decimal address = 4).

The Type filed specifies the format of the values displayed on the telemetry page. We don't have any documentation on this, but I was able to deduce the following:

- Type = 0: integer displayed as decimal
- Type = 1: floating point (uses calibration curve fit)
- Type = 2: integer displayed as Hex
- Type = 3: integer displayed as binary
- Type = 4: Boolean value displayed with label (a label for high, a label for low)

The Mask is used to limit the range of displayed values, which can eliminate "garbage" bits, or bits that are not explicitly set in a telemetry value. For example, if you are downloading a 16-bit value instead of a 32-bit value, you should specify a mask of 0x0000FFFF to prevent the unused higher order bits from being displayed. This prevents erroneous values from being read on the telemetry page.

The min and max values are the absolute minimum and maximum values that can occur for a given telemetry channel (the min and max values that can be reported by the sensor). The low and high values allow the user to specify expected ranges for the telemetry channel values (for nominal operation). If the value is below the min or greater than the max value, that value will display in red (as opposed to black) to indicate to the ground station operator that there is something wrong with that value. If the low and high values are set to –1, it indicates that boundary checking (to display in different colors) is disabled.

The X2,X1, and 1 columns represent the coefficients of a quadratic curve fit for the value. If you let x = value, the actual displayed value will be equal to:

$$\text{Value} = (x2)x^2 + (x1)x + (1)$$

The last high and low columns are the labels that will be used if the data type is set to 4. For a binary high value, it will display the label in the high column. For the binary low value, it will display the label in the low column.

When viewed in Microsoft Excel, a row of the Telecommand CSV file looks like this:

;name	channel	default	min	max	Units	Type	A	B	C	hint
[SBandTx]	4	1	1	0	0	0	0	0		
Set Data source	0x0004000A	0	0	15	-	0	0	1	0	Set data mux

The name column can have several possible settings. As with the telemetry file, a semicolon in front of the "name" value means that this telecommand won't be displayed in the telecommand window. The name in brackets [ ] is the name of the module to which you wish to send the telecommand as shown in the window shown in Figure 28.

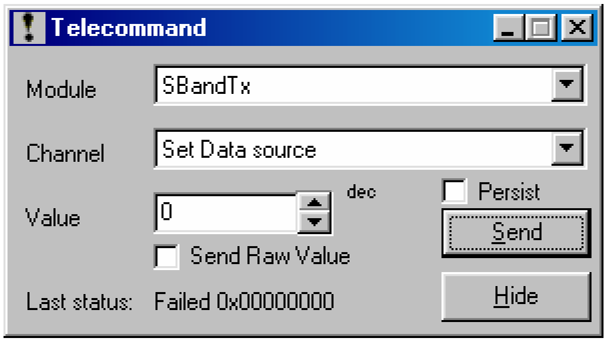


Figure 28: Telecommand dialog window

The [SbandTx] module name is displayed in the module drop list. (the first [] module name is shown by default when you first open up the telecommand window). The names without brackets are the names of possible telecommand channels (which are the actual commands sent to those devices). In this case, Set Data Source is the default channel displayed (because it is the first channel displayed in the .CSV file).

The channel column, as in the telemetry .CSV file is a combination of the CAN address of the module to which the telecommand is being sent (Hex address = 0004, decimal address = 4, which corresponds to the address of the S-Band transmitter), and the telecommand channel (hex value = 000A, decimal value = 10, corresponding to the Set Data Source telecommand). The satellite only thinks of these channels and modules as numbers, so changing the "name" of the channel or module has no impact on what the satellite does. If you want to send a different telecommand, you must change the value in the channel column to make it so.

The default column displays the value you will see by default when you select a particular telecommand channel via the drop list. In this case, the default value for the Set Data Source command is 0 (meaning that the telemetry/telecommand/bootloader responses come from the transmitter – as opposed to the on-board computer, which would have a value of 3 for FalconSat 2).

The high and low values are the range of allowable values for the telecommand. In this case, 0 is the lowest value for data source, and 15 is the highest allowable value that can be sent (for FalconSat2, only the values of 0 or 3 have any meaning, but they are both in the valid range)

The units column allows the user to specify a value in a certain type of units. (For example, we may wish to set a voltage on the satellite, which will automatically convert to the appropriate integer value in counts to send to the satellite). These units will be displayed next to the value window in the telecommand dialog box.

The type column specifies the format of the specified value. The available types include:

- 0 – decimal integer
- 1 – floating point value

Most telecommands are sent as 0s, so we usually will specify this type as 0

A,B, and C are the coefficients in the equation

$$\text{Value} = Ax^2 + Bx + C$$

When type is set to 1, the TLM CAN program will solve for the x that gives the desired value, sending the nearest integer.

Last but not least is the hint column, which is basically a comment about the telecommand.

5.2 FalconSAT-2 Inhibit Switch Verification Equipment

As the name implies, the single purpose of the FS-2 Inhibit Switch Verification Equipment (ISVE) is to verify that all 4 inhibit switches plus the spacecraft separation switch are OPEN while mated to the PES. The ISVE consists of an aluminum box 2" x 3" x 2". All components are internal to the box. Power is provided by an internal 9-V battery. An On/Off switch is located on the top of the box, a GREEN LED provides for "On" indication and verification of internal power. Four RED LEDs, one per switch, provide indication of switch position, OFF = OPEN, ON = CLOSED. These LEDs are labeled on the outside of the box. An electrical schematic of the ISVE is shown in Figure 29.

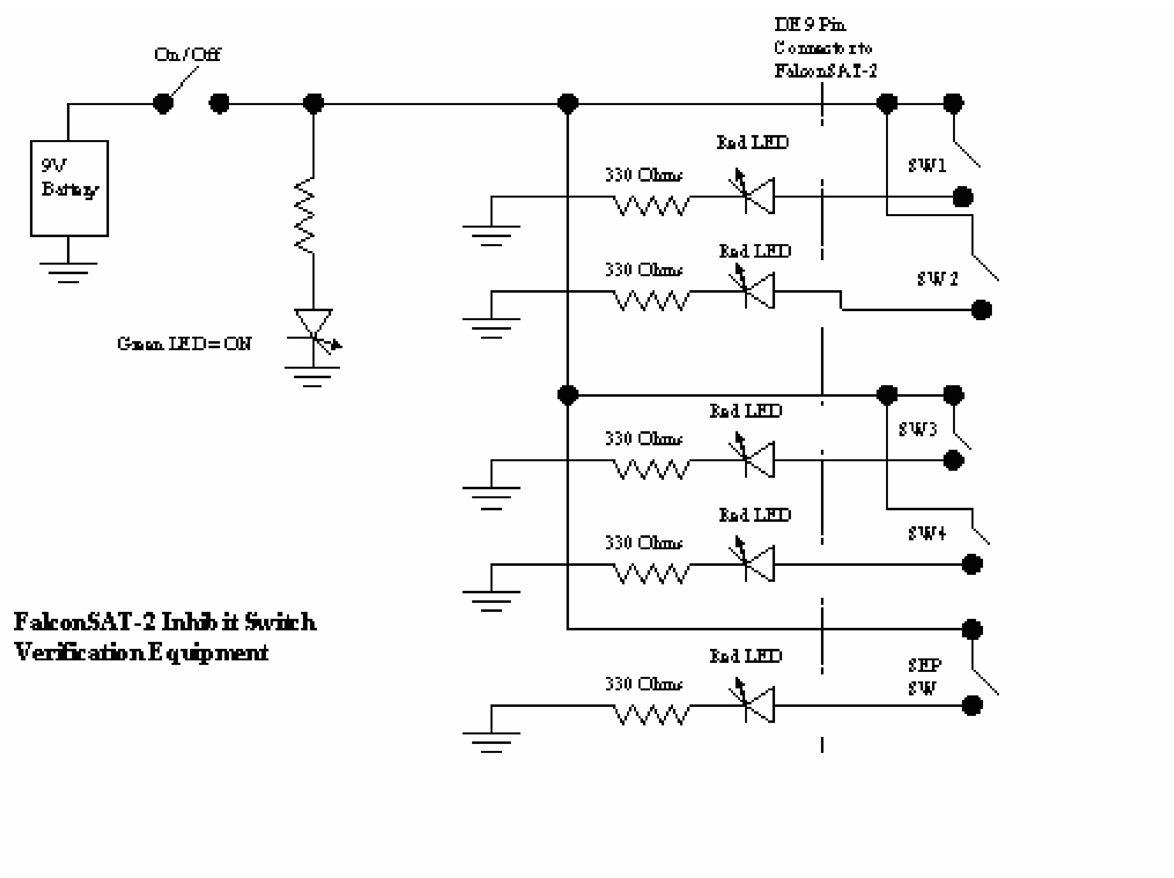


Figure 29: FS-2 Inhibit Switch Verification Equipment Schematic

Operationally, the ISVE is used in the following way:

- All LEDs are OFF when the switch is OPEN
- Insert 9S-9P extension cable into TEST SWITCH DE9P
- Turn switch on, GREEN LED comes on proving battery is good
- Insert test plug
- All RED LEDs should stay OFF and GREEN LED stays ON
- Remove test plug

In addition, FS-2 has built a set of ISVE test plugs to verify the functionality of the equipment. These plugs verify the operation of the LEDs in the ISVE. When inserted into the ends of the ISVE cable they will simulate closed switches. The result should be all 5 red LEDs on the ISVE ON. This proves those LEDs and the cables are working properly. These plugs are built with Continental 6 pin green flat connectors. These connectors have labels of A – F on the pins and are polarized. There is one cable from the ISTB that splits into two cables each with a Continental connector on the end. There is a test plug for each.

The Separation Switch Verification Plug tests the separation switch operation of the ISVE, also called SEP SW. It is wired as shown in Figure 30.

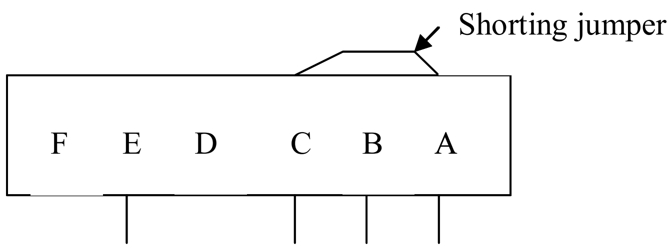


Figure 30: Separation Switch Verification Plug

The Inhibit Switch Plug test the inhibit switch, operation of the ISVE. It is wired as shown in Figure 31.

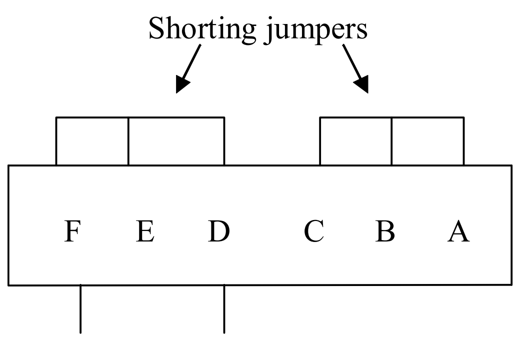


Figure 31: Inhibit Switch Verification Plug

5.3 FalconSAT-2 Battery Charging and re-conditioning Equipment

The FS-2 Battery Charging and re-conditioning Equipment (BCE) will be used periodically from spacecraft delivery to GSFC through launch to provide top-off charging of the flight battery and re-conditioning as necessary. As of this writing, the BCE is not yet fully constructed. However, it will consist of:

1. Fluke 77/BN Multi-meter
2. Kepco model MSK10-10M DC power supply
3. SuperNova 250S Fast-Charger model FC700
4. Spacecraft interface cable with 2-A fuse
5. 110-VAC extension cable (as required)

The Fluke multi-meter will be used separately to measure the state of battery charge prior to performing any procedures. The Kepco power supply, as shown in Figure 32, provides independent voltage and current limits. The SuperNova 250S Fast-Charger, shown in Figure 33, provides "smart" charge and discharge for the battery, providing automatic or manual mode peak-detect battery charger and discharger capable of handling NICAD, NIMH and lead-acid batteries.

The Spacecraft Interface cable connects to the FS-2 Switch Test port on the top of spacecraft as shown in Figure 18. A schematic for the connection BCE is shown in Figure 34. Operational procedures for use of the BCE are given in FS2-BATTPROC-23.

Figure 32: Voltage/Current-limited DC power supply used in FS-2 Battery Charging Equipment.

Figure 33: SuperNova 250S Fast-Charger

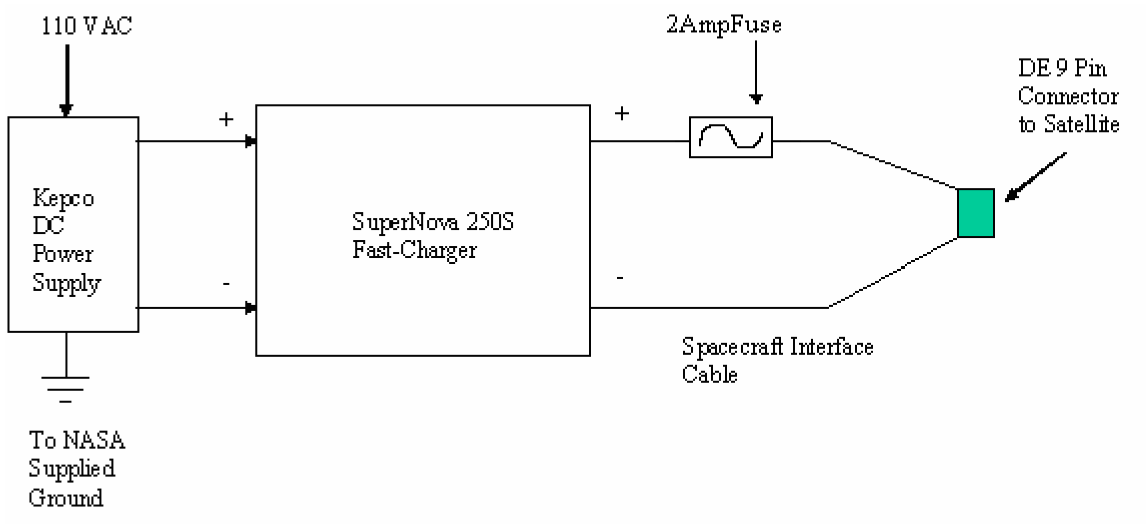


Figure 34: Battery Charging and re-conditioning Equipment (BCE) Schematic

5.4 Battery charge/discharge procedures

It is important to confirm the state of the battery at the beginning of charge. This requires that the battery be fully discharged prior to charging using the following method. The SUPERNOVA 250S charger is programmed to provide this discharge cycle prior to charging. It will discharge the battery at a C/5 rate (800 mAmps) until the voltage of the battery drops to 1 volt per cell. At this point, the SUPERNOVA stores the data on the discharge and begins the charge cycle.

1. Instruction for use of the SUPERNOVA 250S charger
   - Power supply
     Use a DC power source of 12V and at least 10A
     Confirm voltage setting with the multimeter before connecting the charger
   - Steps for using the SUPERNOVA to complete discharge/charge cycle
     a. _connect the charger to the power supply and press SET/STOP to enter the main menu
     b. _select the “battery” menu by pressing UP or DOWN until BATTERY is flashing then press SET/STOP
     c. _SELECT must be flashing so press SET/STOP to enter the “select” menu. Select the needed battery setting (NiCd, 7 cells, C=4200, Cc=0.4A, Dc=0.8A) with UP or DOWN then press SET/STOP
     d. _press UP or DOWN until exit is flashing, then press SET/STOP to reach the main menu
     e. _press DOWN and SET/STOP to enter “start”
        -- this is the manual start menu and the selected battery’s characteristics are flashing
     f. _ connect the battery to the charger
     g. _press the DISCHARGE button to begin a discharge/charge cycle.
     h. _The charger may take up to 3 minutes to check the battery prior to beginning the cycle
     i. _Once the voltage is under 7V the SUPERNOVA charger will automatically start charging the battery.

j. _The SUPERNOVA charger will automatically stop charging when battery is fully charged. (See SUPERNOVA manual for details on peak charge sensing.)
k. _Charge for no more than 16 hours.

APPENDIX A: ACRONYMS

ADCS	Attitude Determination and Control System
BCE	Battery Charging Equipment
BCR	Battery Charge Regulator
BPF	Band pass filter
bps	Bits per second
BPSK	Binary Phase Shift Key
C&DH	Command and Data Handling Subsystem
CAN	Control Area Network
CARS	Customer Accommodations and Requirements Specification
CGSE	Customer Supplied Ground Support Equipment
COMM	Communication Subsystem
CONUS	Continental United States
COTS	Commercial off-the-shelf
DoD	Department of Defense
EDAC	Error Detection and Correction
EMC	Electromagnetic Compatibility
EMI	Electromagnetic Interference
EOM	End of Mission
EPROM	Erasable Programmable Read Only Memory
EPS	Electric Power Subsystem
FOV	Field of View
FPGA	Field Programmable Gate Array
FS-2	FalconSAT-2
FSDP	Flight Safety Data Package
FSK	Frequency Shift Keying
FTE	Functional Testing Equipment
GaAs	Gallium Arsenide
GFE	Government Furnished Equipment
GSE	Ground Support Equipment
GSFC	Goddard Space Flight Center
HDLC	High-level Data Link Controller
HH	Hitchhiker
HMDA	Hitchhiker Motorized Door Assembly
IBF	Insert before flight
IF	Intermediate Frequency
ISVE	Inhibit Switch Verification Equipment
JSC	Johnson Space Center
KSC	Kennedy Space Center
LA	Laminated Analyzer
LEO	Low Earth Orbit
LNA	Low noise amplifier
LO	Local Oscillator
LPF	Low pass filter
MESA	Miniature Electrostatic Analyzer

MET	Mission elapsed time
MIB	MESA Interface Board
MIPS	Millions of instructions per second
NASA	National Aeronautics and Space Administration
NiCd	Nickel-Cadmium
NSTS	National Space Transportation System
NTE	Not To Exceed
OBC	Onboard Computer
OTP	One Time Programmable
PCB	Printed circuit board
PCM	Power Conditioning Module
PDM	Power Distribution Module
PES	Pallet Ejection System
PLB	Payload Bay
POCC	Payload Operations Control Center
PPF	Payload Processing Facility
RAM	Random Access Memory
RBF	Remove before flight
RCS	Reaction Control System
RF	Radio Frequency
RISC	Reduced Instruction Set Computer
ROM	Read Only Memory
RPA	Retarding Potential Analyzer
SERB	Space Experiments Review Board
SIM	Subsystem Integration Module
SNAP	Surrey Nanosatellite Applications Platform
SSTL	Surrey Satellite Technology Ltd.
STPI	Satellite Test Port Interface
TNC	Terminal Node Controller
UART	Universal Asynchronous Receiver Transmitter
USAF	United States Air Force
USAFA	United States Air Force Academy
VHF	Very High Frequency

APPENDIX B: PAYLOAD UNIQUE HAZARD REPORT

- FS2-G-01 Electrolyte Spillage, Rupture of Battery, or Ignition of Hazardous Atmosphere

PAYLOAD HAZARD REPORT		NO: FS-2 G-01
PAYLOAD: FalconSat-2		PHASE: 2/3
SUBSYSTEM: FS-2 Electrical	HAZARD GROUP: Explosion	DATE: 24 May 2002

HAZARD TITLE:

Electrolyte Spillage, Rupture of Battery, or Ignition of Hazardous Atmosphere

APPLICABLE SAFETY REQUIREMENTS:	HAZARD CATEGORY:
KHB 1700.7B, para.: 4.3.2 Electrical; 4.3.9 Electrical Systems; 4.4.2 Hazardous Atmosphere	X CATASTROPHIC
NSTS 20793	CRITICAL

DESCRIPTION OF HAZARD:

Explosion of NiCd battery, release of corrosive agent due to rupture of the battery, or ignition of flammable atmosphere.

HAZARD CAUSES:

1. Excessive accumulation of battery gas during battery charging due to overcharge.
2. Incorrect connection of polarity between battery GSE and battery.
3. Ignition of flammable atmosphere due to improper GSE operation.

HAZARD CONTROLS:

1.1 Battery top charge regulator provides monitoring of cell voltage to identify end of charge.
1.2 Battery top charging will be performed according to approved procedures.
2.1 EGSE connection procedures will specify use of keyed connectors to prevent mismating.
3.1 Battery charging GSE will be powered at a monitored source with a facility provided master kill switch.

SAFETY VERIFICATION METHODS:

1.1.1 Design review. GSFC Code 743 to review and concur with USAFA Battery Charging Equipment design.
1.2.1 Procedure review. GSFC Code 743 to review and approve battery charging procedure.
2.1.1 Inspection. GSFC Code 743 to inspect GSE to insure connectors are properly sized and keyed.
2.1.2 Procedure review. GSFC Code 743 to review and approve charging procedure to ensure proper GSE connection and operation.
3.1.1 Procedure review. GSFC Code 743 to review and approve procedure to verify use of launch pad monitored power source.

STATUS OF VERIFICATION:

1.1.1 Closed to VTL.
1.2.1 Closed to VTL.
2.1.1 Closed to VTL.
2.1.2 Closed to VTL.
3.1.1 Closed to VTL.

APPROVAL	PAYLOAD ORGANIZATION	STS

PHASE I		
PHASE II		
PHASE III		

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